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AVIONIC COOLING AND POWER SUPPLIES FOR ADVANCED AIRCRAFT.(U)
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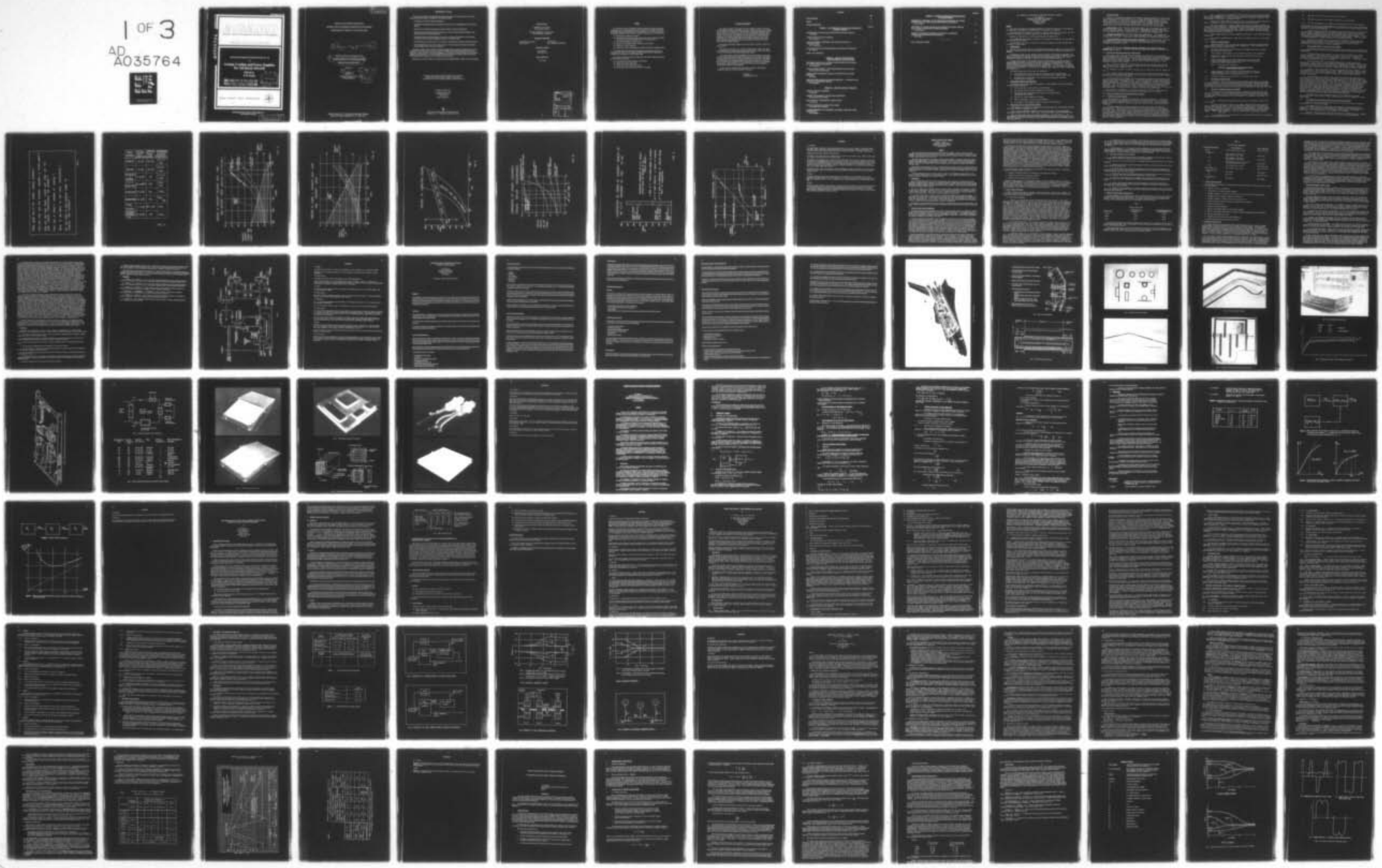

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Avionic Cooling and Power Supplies for Advanced Aircraft

Edited by
P.W.Smith

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AVIONIC COOLING AND POWER SUPPLIES FOR ADVANCED AIRCRAFT.

Edited by

10

P.W. Smith

Ministry of Defence (PE)
United Kingdom

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THEME

The continued increase in the quantity of avionics equipment in military aircraft has already given rise to a critical situation in terms of cooling. The environment, particularly at high speed and low level has made the use of the airframe or the fuel as a heat sink a less profitable arrangement than in the past. Alternative solutions must be found including:

(a) Reduction in the quantity of avionics

(b) More efficient use of primary power (e.g. in some aircraft the increase in demand of 1 Kw for electronics is equivalent to 50 Kw extra engine power)

(c) More efficient cooling of avionics systems

(d) Increase in the acceptable ambient temperature of components

(e) Reduction in critical components

(f) Reduction in copper conductors and more efficient distribution of generated heat.

The airframe industry has already given presentations expressing concern over the serious situation, which, if uncontrolled could seriously inhibit the proper applications of valuable avionic technology to the total weapon design.

The purpose of this meeting will be to disseminate information outlining the problems and likely solutions in order to:

(a) Alert system and airframe designers to the problems

(b) Define the problems quantitatively

(c) Identify areas which require urgent research

(d) Advise those concerned of the present state of knowledge.

EDITOR'S FOREWORD

The continued increase in the quantity of avionics equipment of which an increasing amount is digital in operation, has already given rise to a critical situation in terms of cooling and the provision of electrical power. The environment, particularly at high speed and low levels, has made the use of the airframe as a heat sink a less profitable arrangement than in the past. At the same time the introduction of a new generation of computers has made possible their use in flight critical systems and new system areas such as Active Control Technology. The use of these high integrity systems together with the increase in total avionic equipment makes demands on the electrical supply and utilization which cannot easily be met with existing equipment.

This specialist meeting has confirmed the need for alternative solutions to these interacting problems.

The sixteen papers presented to an audience of eighty delegates, ranged from a paper on the technology of heat pipes to papers reviewing the problem as a whole. This report is a record of these papers and the lively discussions which ensued.

In his closing address the Chairman highlighted the need for in depth avionic cooling and power supply trade-off-studies, related to the overall role of the aircraft. These studies he said, would point the way to co-ordinated programmes of research which would bridge the interface between the equipment suppliers and the aircraft constructors. In financial terms the possible cost savings quoted in the paper entitled "The effect of Avionics System characteristics on fighter aircraft size, cooling and electrical power supply Subsystems," would justify the need for such a programme.

He then went on to thank all those whose efforts in support of this specialists meeting were so generous and had contributed to its success.

P.W.SMITH
Programme Editor 1976

CONTENTS

THE PROBLEMS OF COOLING HIGH PERFORMANCE MILITARY AIRCRAFT

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SUMMARY

This paper discusses the problems imposed on the aircraft designer in disposing of rejected heat, and examines the penalties on the aircraft of doing so.

The reasons why the problem is so great in today's generation of aircraft, and the probability of growth in the next generation of aircraft are discussed.

Sources of heat and their cooling requirements are identified.

It is shown that the quantity of heat to be rejected has increased, but the mass of the aircraft has decreased.

In particular, the effect of avionic cooling requirements on total aircraft heat rejection is illustrated.

1. INTRODUCTION

This paper illustrates some of the problems which present themselves to the aircraft designer in rejecting heat, and the penalties on the aircraft design and performance in doing so.

Although the theme of this meeting concerns avionics and electrical equipment, in order to illustrate the cooling problem, it is necessary to initially broaden the subject so that the particular problems of cooling avionics equipment are put in the correct perspective relative to the other aspects of the overall weapons system.

I believe that many of the problems which are encountered in cooling avionics in today's high performance aircraft have arisen because of long-standing traditional 'compartmentation' of the subjects of avionic and mechanical engineering. This has been true both in Ministerial Departments and in Industry. At last there is evidence of more 'across the board' awareness of the problems, and I trust that this meeting and this paper's contribution to it will serve to improve the situation further by extending that awareness.

The major objectives of the paper are therefore:-

i. to show that heat loads, relative to the size of the aircraft have increased over recent years, and future trends show a similar growth rate.

ii. to illustrate the magnitude of the aircraft and equipment penalties caused by heat rejection.

2. THE COOLING PROBLEM AND PENALTIES

In dealing with the main body of this paper, I will endeavour to provide answers to the following questions:

a) What gives rise to the overall cooling problem?

b) How and why has the problem changed in recent years?

c) How are the heat loads cooled and what are the main factors compounding the problem?

d) How big is today's penalty of cooling?

e) What is the effect of the local environment?

f) Do we need to take positive steps to improve the situation?

2.1 What gives rise to the cooling problem?

This question is answered by identifying and quantifying the various heat sources, with a few words of explanation of each.

Fig. 2 gives a tabulated summary of the main heat sources in a modern high performance military aircraft, their effect and the necessary rejection temperatures.

Only two basic heat sinks are normally available:- fuel and ram air. The application of these heat sinks varies considerably from aircraft to aircraft, but the heat sink shown in the last column of this slide represents the most frequently used in military aircraft.

Kinetic Heating

Kinetic (or aerodynamic) heating contributes a significant quantity to the overall heat balance of cabin and equipment bays. Aircraft skin temperatures of 100°C or more can be encountered in "below radar" low level flight at transonic speeds, and even higher temperatures are reached in supersonic operation at medium and high altitude. It should also be noted that in some flight cases, (e.g. subsonic cruise at altitude), kinetic heat loads can be negative, although very often this does not give the alleviation to the overall cooling problem which one may at first sight imagine.

Solar Heating affects the cabin heat load directly and the equipment bay heat load indirectly during flight. This is particularly emphasised by the requirements for large transparent canopies on modern aircraft. During ground standby, solar heating is a very significant factor, affecting both the cabin and the equipment bays.

Avionics and Radar heat loads are largely self explanatory. The significant factors here are the efficiency of the avionic equipment and the effectiveness in using the available heat sink. More will be said about these very important topics later.

ECS (Environmental Control System) heat rejection results from tapping air at very high temperatures from the engine compressor, and having to cool this air by heat exchangers and expansion turbines to provide a suitable temperature for cabin and equipment environmental control.

Heat loads imposed by Hydraulic systems, Generators, Fuel Pumps, Engine and Gearbox oils are all self explanatory and present significant portions of the overall aircraft heat balance.

2.2 How has the problem changed in recent years?

Over recent years, the necessity for aircraft to carry more, and more complex avionic equipment has increased. This is brought about mainly by the rapid development of new techniques in modern warfare, and the trend towards more electronic and computer-aided aircraft and engine controls. The other very significant factor is that at the same time, aircraft mass has reduced, and is likely to reduce even further. What evidence is there for this claim?

Fig. 3 shows the trends of total aircraft heat load and aircraft mass over the last 20 years or so. This data has been compiled from a fairly broad selection of American and European combat aircraft.

The divergent situation of smaller aircraft with a larger heat load can clearly be seen.

Fig. 4 shows a clearer illustration of the trend in cabin and avionic heat loads, which are combined together here because in most aircraft using air for avionics cooling, the two are very much inter-dependent.

It has frequently been claimed that the development of solid state circuitry would reduce aircraft and equipment heat loads. I have not been able to find any evidence to support these claims. Although the heat load per component has reduced, this has been greatly outweighed by the increase in the number of components. In fact, the situation has arisen where, with solid state circuitry the heat load per unit volume has actually increased. Fig. 5 illustrates this point by comparing the heat load and volume of equipment having similar functions. Similar claims of future reduction of heat loads are being made today. I would very much like to see evidence to support them.

In most avionic equipments, a very large proportion of heat rejected comes from the power supplies, which give the necessary voltage control for the device.

2.3 What are the main factors compounding the problem?

2.3.1 Trends in avionics design

The necessity for improved reliability is very well understood. In fact the reliability achieved is probably one of the most significant factors which determines whether a particular weapons system is a good one or not so good! The relationship between individual component temperature level and reliability is also understood and documented.

In many instances the conclusion appears to have been drawn that improved reliability of a complete piece of equipment can be achieved simply by increasing its cooling airflow. This may be true in a few cases, but unfortunately in many others it is not. The effectiveness of the cooling methods employed in black box design does vary considerably, and in many examples of equipment in use today, a large increase in cooling airflow will give only a small reduction in component temperature, even when the cooling air temperature is well below the temperature of the components being cooled. This is because of the low effectiveness of the heat transfer techniques used in the design of these equipments.

Fig. 6 illustrates the magnitude of the variation in cooling airflow demands for a range of modern avionic black boxes, all of which have similar components doing similar types of function. It can be clearly seen that a factor of 3 exists in mass flow demands for the range of equipment designs.

Although the ARINC standards basically apply to commercial aircraft avionics, there is frequently a read-across from commercial to military avionics.

The early standard of cooling to ARINC 404 required 0.5 lb/min (at 38°C) per 100 watts of heat dissipation. Miniaturisation and increased package density has led to more air being required in the ARINC 404A standard - this being 0.8 lb/min (at 38°C) per 100 watts to try to improve reliability. Recent developments with high density digital electronics have increased the requirements still further, and much equipment now demands 1.0 lb/min per 100 watts.

These figures apply for avionic equipment installed in a very moderate ambient temperature.

The current recommendation of ARINC 600 is 218 kg/hr per kW (i.e. 0.8 lb/min per 100 watts).

2.3.2 Trends in engine design

Engine development over the years has resulted in multispool engines giving higher specific thrust and lower specific fuel consumption. This has added to the overall cooling problem in several ways:-

(a) additional bearings and higher turbine entry temperatures have increased the engine heat rejection.

(b) large capacity reheat pumps require cooling in reheat 'Off' conditions.

(c) reduced fuel flow means less available heat sink directly into the engine fuel supply.

(d) higher engine compressor delivery temperature results in an even lower environmental control system Co-efficient of Performance, resulting in higher engine thrust loss.

NOTE: Coefficient of Performance

$$= \frac{\text{Effective refrigeration obtained from system}}{\text{input energy to system}}$$

(e) higher compressor bleed flows have a more detrimental effect on engine handling. (For a high bypass ratio engine, the bleed is a higher percentage of the engine core flow.)

(f) Electronic engine control units add to the avionic heat load.

2.3.3 Trends in airframe design

The smaller aircraft has resulted in much tighter packaging of all systems, including avionics. This close packing means that heat losses from black box cases by radiation and natural convection are virtually non-existent. Where black boxes rely on these modes of cooling, additional airflow must be provided along the sides of the boxes, to assist this heat loss, and prevent transmission of heat from box to box.

2.4 How big is today's penalty of cooling?

The main penalty of cooling on aircraft performance results from the heat rejected to ram air. The environmental control system, which conditions the cabin and cools the avionics is by far the largest single power consumer on the aircraft. The magnitude of the problem can be well illustrated by considering the penalty of a typical ECS.

Fig. 7 illustrates the energy used by typical ECS designed to cool a heat load of 30kW.

It can be seen that in order to cool 30kW, approx.300kW of engine power is used to supply the ECS airflow, and a further 400kW of engine power is required to overcome the aerodynamic drag of the ECS (mainly heat exchangers.) Thus for every 1kW increase in avionic heat load, today's cooling methods demand that nearly 25kW of power is drained from the engines - and this is considered to be a very efficient system. (NOTE: This represents an ECS true coefficient of performance of less than .05)

It is interesting to note also that in typical modern aircraft, the illustrated penalty is approximately equal to -

(i) The drag of a 1000 lb bomb carried on the aircraft

or

(ii) 10 miles reduction in the radius of operation of the aircraft

or

(iii) 10 miles penetration of an enemy intruder before the point of interception.

In addition to the total performance penalty on the aircraft, we must also consider the mass and bulk of the equipment which has to be installed in the aircraft purely for the purpose of cooling other equipment.

Today's aircraft, although much smaller than their predecessors, require much more space for cooling equipment. We have much larger pipes, and, because of the greater number of equipments requiring direct cooling, more of them. We have more valves, cooling turbines and above all, more larger, heavier heat exchangers. We are now seeing heat exchangers weighing more than 30 kg being installed. The total mass of cooling equipment in a modern aircraft can be as much as 300 kg.

Because of the installation and structural constraints of designing these large items into the aircraft, it is seldom feasible to achieve anything like the theoretically possible thrust recovery from used cooling air, and the use of regeneration techniques becomes virtually impossible.

2.5 What is the effect of the local environment?

In discussing this question we will consider the effects of the environment on the aircraft as a whole, and look at the potential savings by designing to lower ambient temperatures.

For the purpose of this exercise, we will examine the effects on a combat aircraft operating at transonic speeds at sea level. First of all, consider the effects in flight. Fig. 8 shows the relationship between static (or ambient) temperature and ram temperature for a constant Mach number of 0.9. Temperature limits for various geographical areas are also shown.

Of most interest here is the relationship between ECS airflow and ram temperature. This is not a straight-forward relationship, because many other aspects, in particular humidity, have to be considered in computing such a curve. However, the main point to be illustrated is that quite drastic limitations on operating temperature of the aircraft would have to be imposed to achieve any worthwhile saving in ECS flow.

Perhaps of more significance, is that by designing the ECS to operate at lower ram temperatures, a lower overall pressure ratio together with slightly larger turbine nozzle area of the cold air unit could be used. This could give up to 10% more flow in engine idle cases with some (although probably very marginal) improvement in avionics reliability.

When operating on the ground, with cooling for the avionics being ambient air induced by fans, a reduction of ambient temperature from 50°C to 30°C would result in the cooling airflow demand being reduced by approximately 25%. The mass of the cooling fans in comparison to the overall ECS may be very low and so very little overall saving in aircraft weight would be achieved. However, it is generally this ground operating case which designs cold wall pressure loss. Designing cold walls with effectively higher pressure loss for the same mass flow (equivalent to the same pressure loss at lower mass flow), could give rise to more effective heat transfer within some black boxes (again with marginally improved reliability), or reduced flow requirements. Because of the vast variation in cooling effectiveness of avionic equipment, it is very difficult to be quantitative with any real meaning. However, it is unlikely that any dramatic improvements would be made unless the overall effectiveness of the cooling methods employed in many of the black boxes was greatly improved.

In addition to all the immediate technical (and commercial) arguments, consideration must also be given to future sales potential of the aircraft, in a world wide context.

2.6 Do we need to take positive steps to improve the situation?

From the brief evidence presented so far it is very clear that the answer to this question is a very positive <u>YES</u>.

Development work both in the avionic and mechanical engineering fields is necessary to reconcile the increasing demand for cooling with the limited ability to provide it. Exact definition of this development work requires a much deeper examination of the problems than has been possible in this paper.

However, I trust that this paper has achieved its intended purpose of creating a true awareness of the magnitude and urgency of reducing the cooling problem.

What gives rise to the overall cooling problem ?

How and why has the problem changed in recent years ?

How are the heat loads cooled and what are the main factors compounding the problem ?

How big is today's penalty of cooling ?

What is the effect of the local environment ?

Do we need to take positive steps to improve the situation ?

FIG. 1.

HEAT SOURCE	TYPICAL HEAT LOAD -kW	APPROX. TEMP. LEVEL - °C	PROBABLE COOLING METHOD
KINETIC	-5 TO 15	30/70	E.C.S. AIR.
SOLAR	2 TO 6	30/70	E.C.S. AIR
AVIONICS + RADAR	5 TO 20	70	E.C.S. AIR.
E.C.S. REJECTION	50 TO 400	150	RAM AIR
HYDRAULICS	10 TO 20	135	FUEL
GENERATORS	10 TO 20	150	FUEL/AIR
FUEL PUMPS	5 TO 30	80 - 150	FUEL
ENGINE + GEARBOX OIL.	15 TO 150	175	FUEL.

FIG. 2

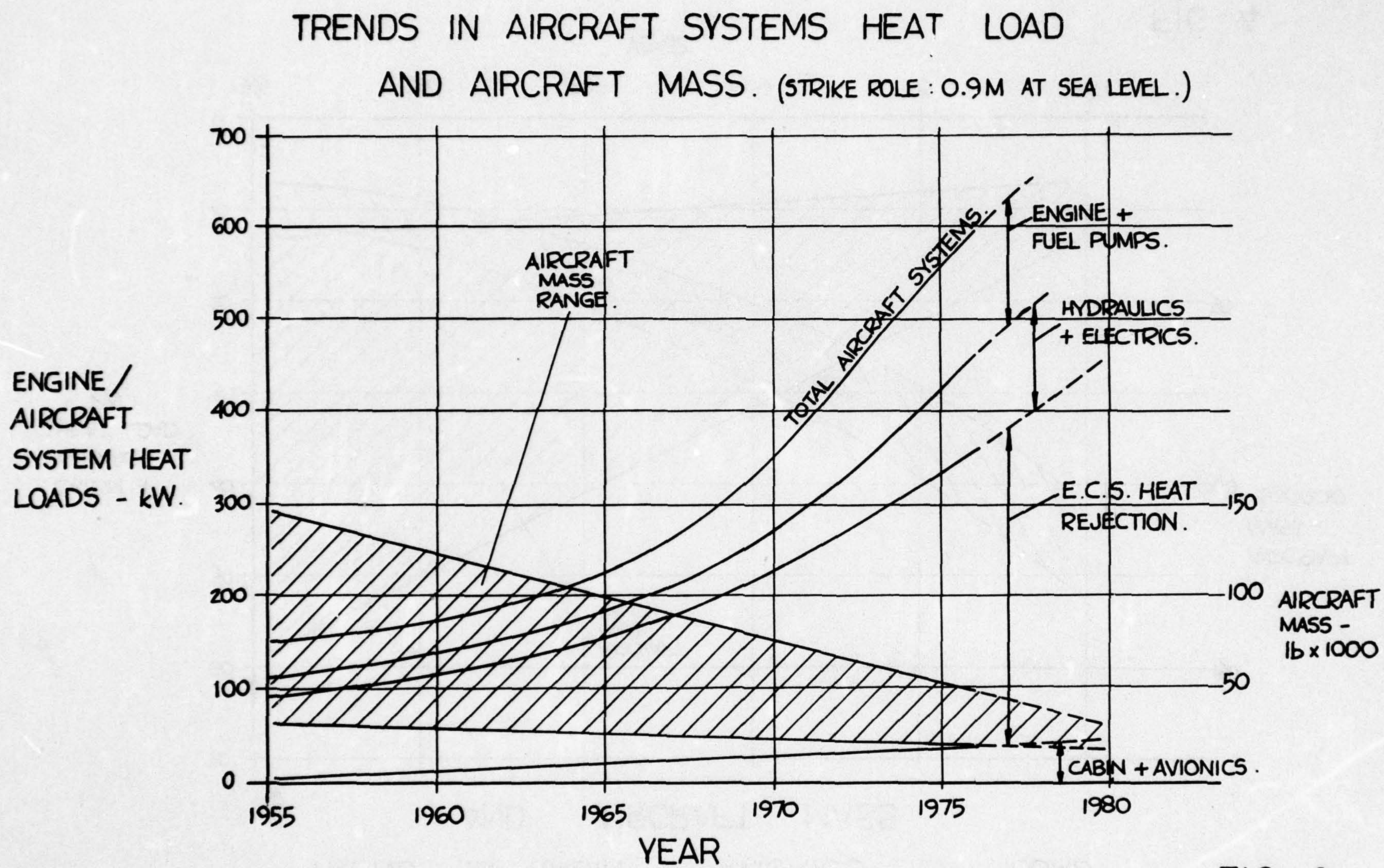

TRENDS IN AIRCRAFT SYSTEMS HEAT LOAD
AND AIRCRAFT MASS. (STRIKE ROLE : 0.9M AT SEA LEVEL.)
ENGINE / AIRCRAFT SYSTEM HEAT LOADS - kW.
700
600
500
400
300
200
100
0
1955
1960
1965
1970
1975
1980
YEAR
AIRCRAFT MASS RANGE.
TOTAL AIRCRAFT SYSTEMS.
ENGINE + FUEL PUMPS.
HYDRAULICS + ELECTRICS.
E.C.S. HEAT REJECTION.
CABIN + AVIONICS.
150
100
50
AIRCRAFT MASS - lb x 1000.

FIG. 3.

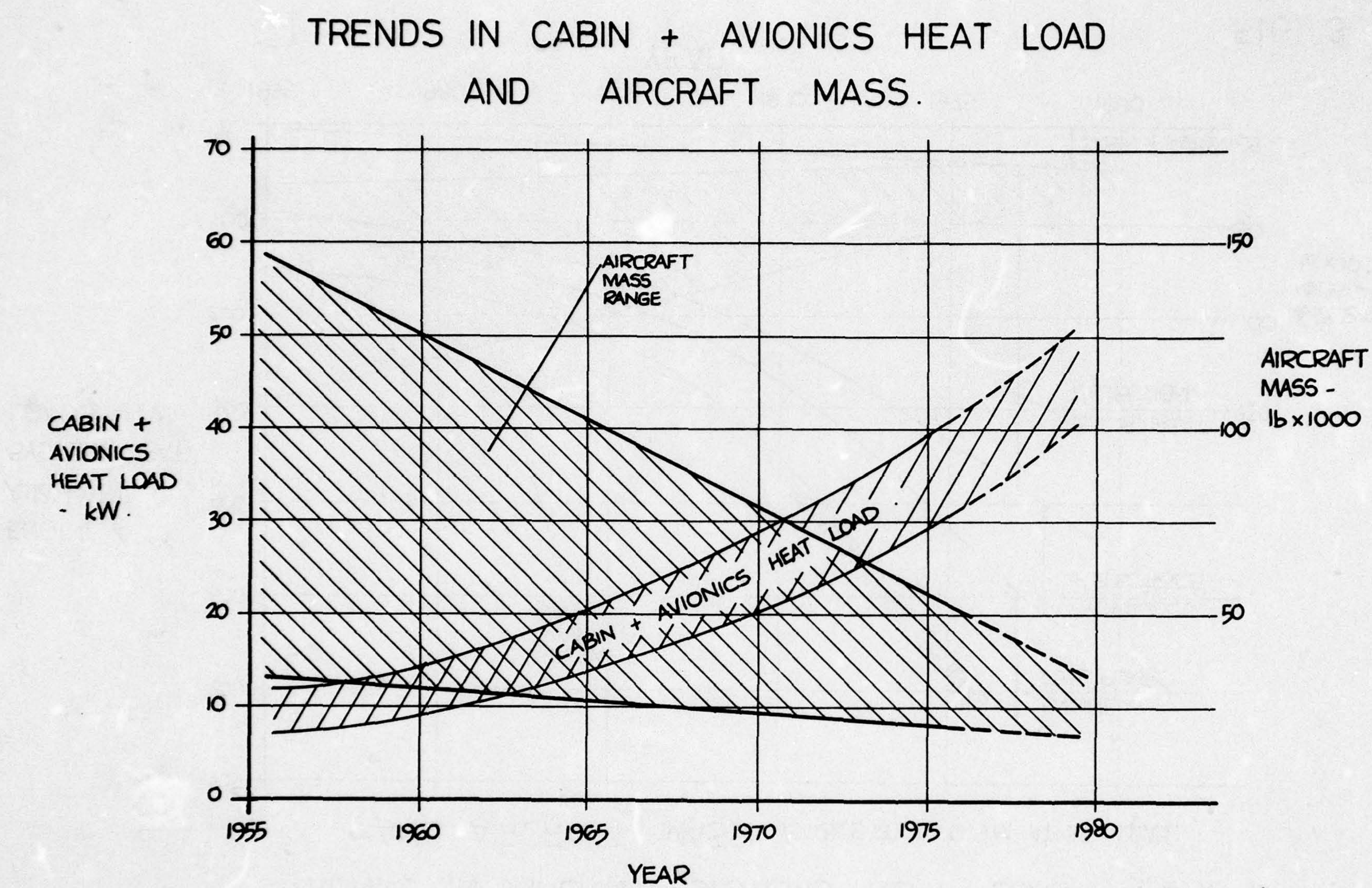

TRENDS IN CABIN + AVIONICS HEAT LOAD
AND AIRCRAFT MASS.
CABIN + AVIONICS HEAT LOAD - kW.
70
60
50
40
30
20
10
0
1955
1960
1965
1970
1975
1980
YEAR
AIRCRAFT MASS - lb x 1000
150
100
50
AIRCRAFT MASS RANGE
CABIN + AVIONICS HEAT LOAD

FIG. 4.

TRENDS IN COMPUTER DESIGN.

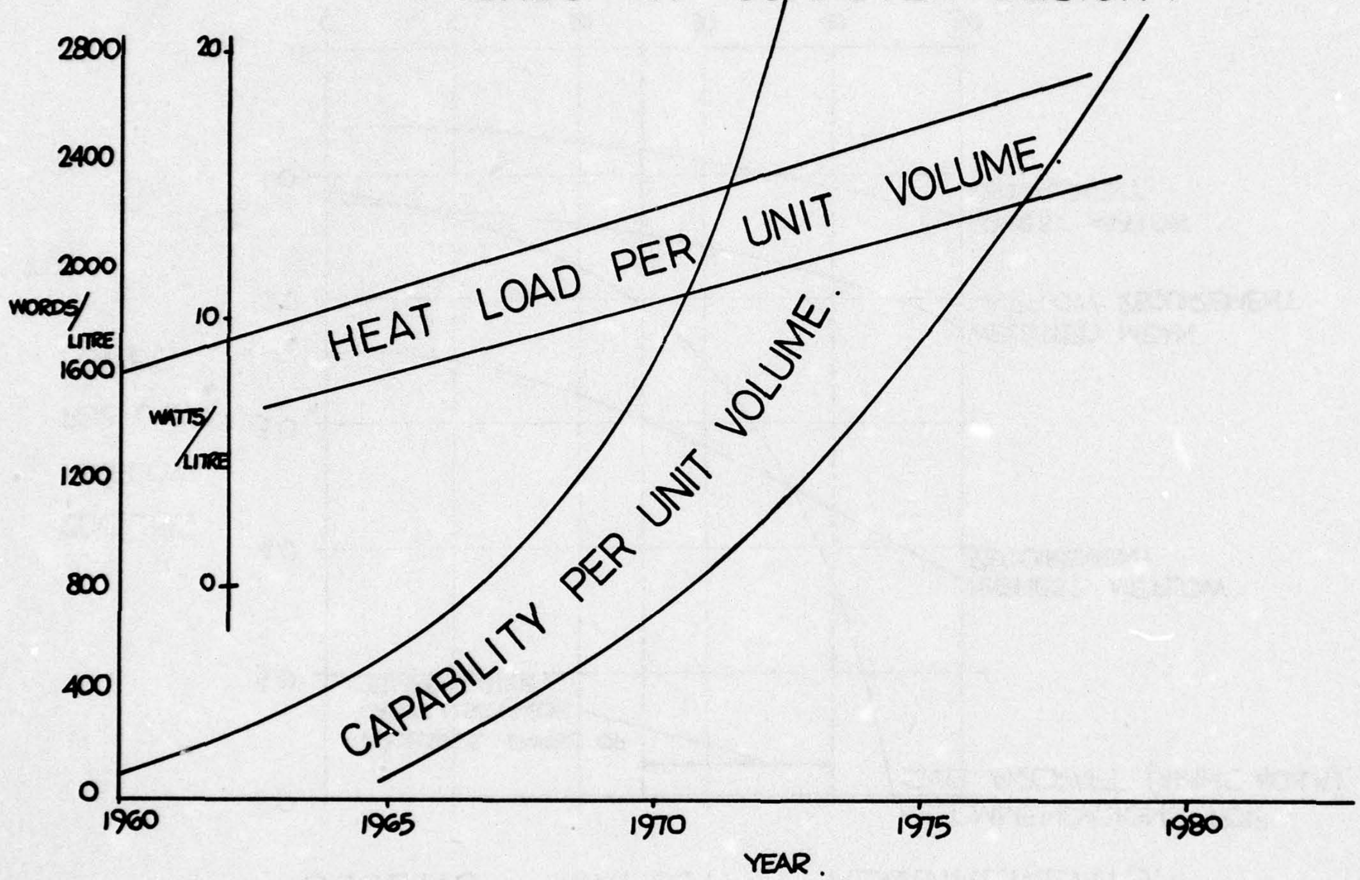


FIG. 5.

COMPARISON OF AVIONIC EQUIPMENT COOLING AIRFLOW REQUIREMENTS.

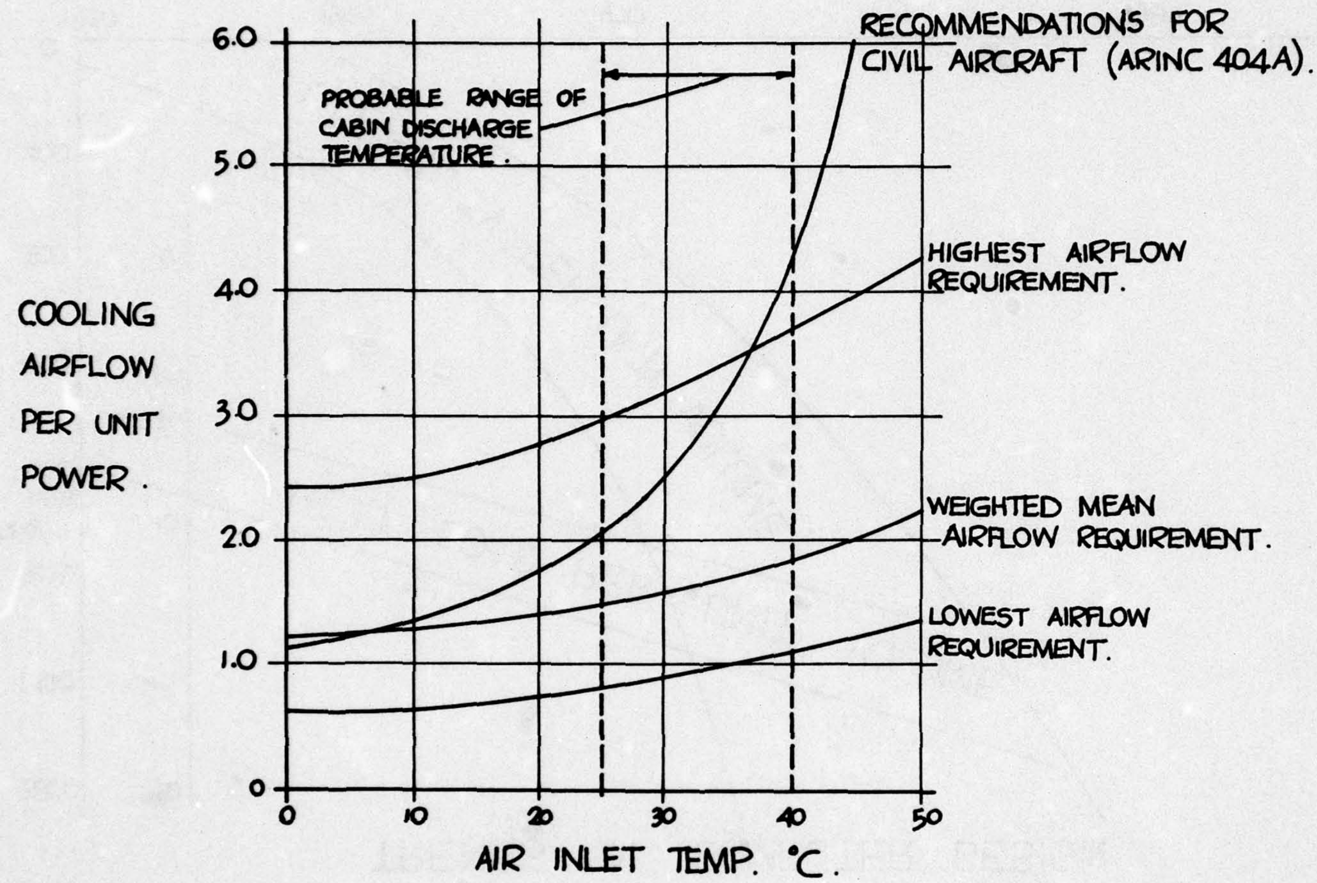
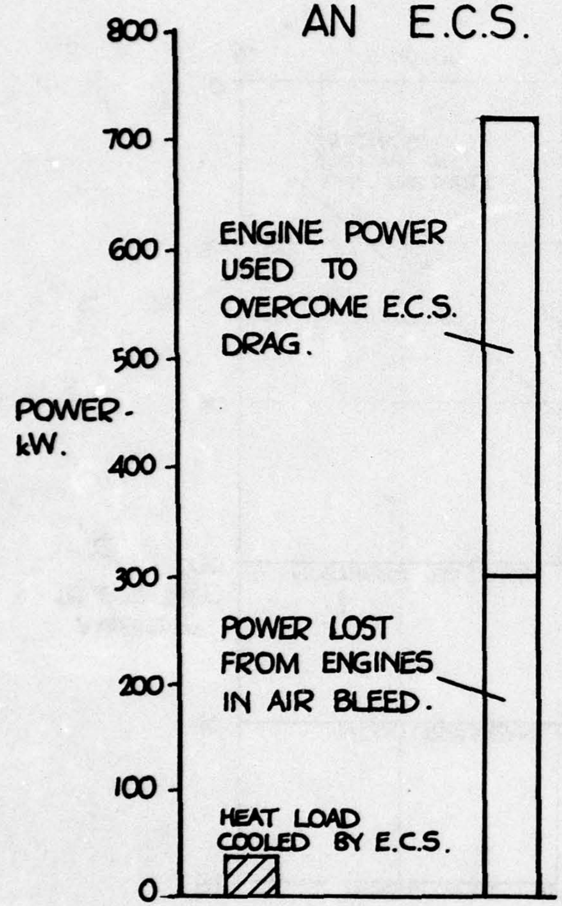


FIG. 6.

ILLUSTRATION OF ORDER OF AIRCRAFT PENALTY OF AN E.C.S. DESIGNED TO COOL 30 kW.



APPROXIMATE EQUIVALENTS OF E.C.S. PENALTY
(TYPICAL VALUES - NOT RELATED TO ANY SPECIFIC AIRCRAFT.)

1. DRAG OF A 1000 lb BOMB CARRIED ON AIRCRAFT.

OR 2. 10 MILES REDUCTION IN RADIUS OF OPERATION.

OR 3. 10 MILES OF PENETRATION OF INTRUDER BEFORE POINT OF INTERCEPTION.

FIG. 7.

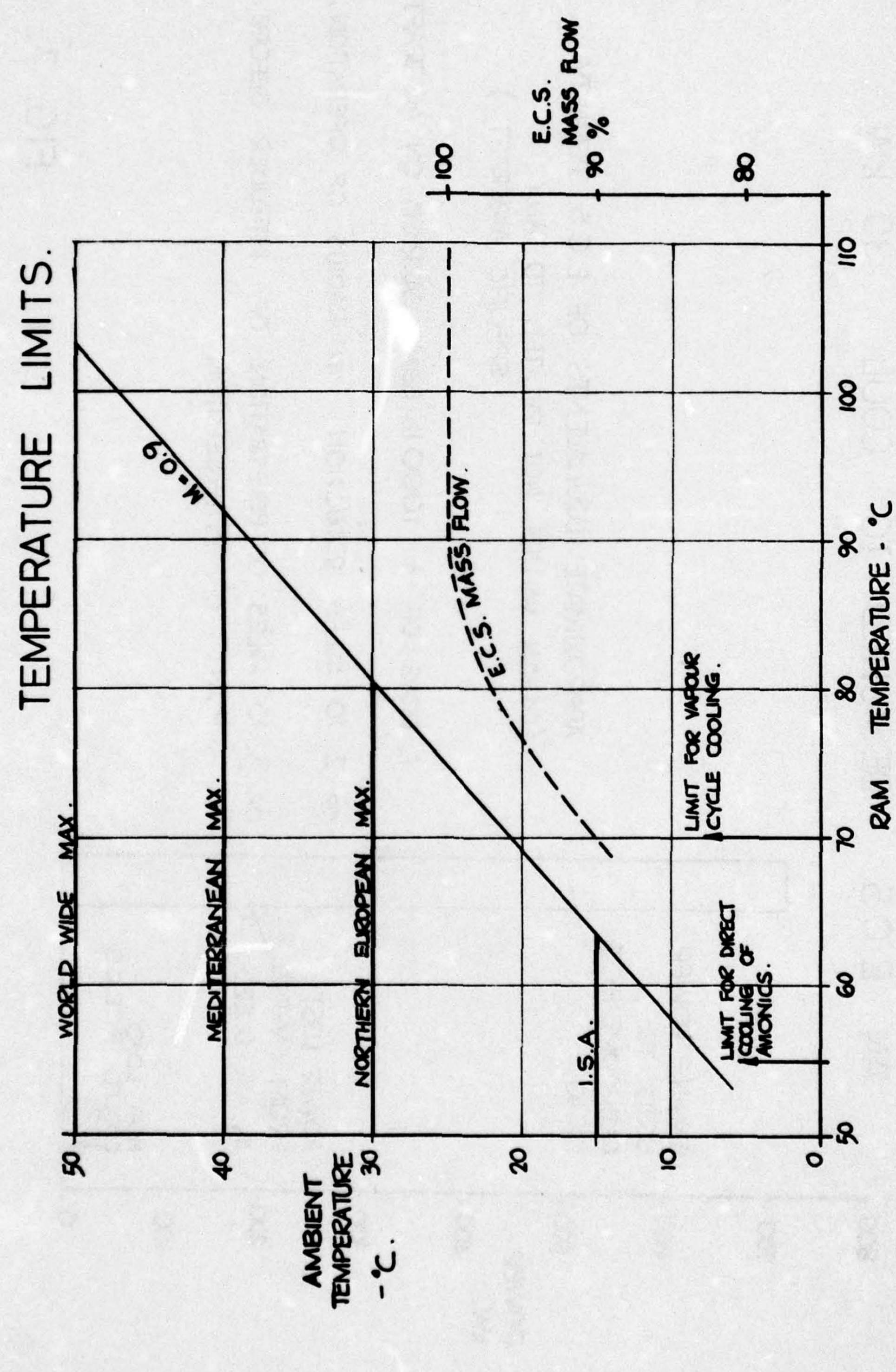

TEMPERATURE LIMITS.
AMBIENT TEMPERATURE - °C.
RAM TEMPERATURE - °C
E.C.S. MASS FLOW %
WORLD WIDE MAX.
MEDITERRANEAN MAX.
NORTHERN EUROPEAN MAX.
I.S.A.
M-0.9
E.C.S. MASS FLOW
LIMIT FOR DIRECT COOLING OF AVIONICS.
LIMIT FOR VAPOUR CYCLE COOLING.
0
10
20
30
40
50
60
70
80
90
100
110
80
90
100

FIG. 8

DISCUSSION

F S Stringer:

You have taken the example of a 30kW load and stated that the load on the engine is effectively 300kW and 400kW. (ECO). Are these to be added directly to obtain an effective overall load of 700kW. If the Avionics load is reduced (say by 50%), would these be a linear reduction in the engine load penalty?

K Morgan, BAC Warton (who read paper on behalf of author)

The answer to the first part of the question is yes, in that for the example stated, 700kW of engine power are required to provide cooling for an ECS of 30kW.

The answer to the second part of the question is that there would not be a linear reduction in the engine load penalty if the Avionics load only were reduced by say 50%. It was stated in the paper that for the purpose of this particular study the cabin and Avionic heat loads had been added together to form the ECS as they were very much inter dependent. For the flight case considered in the example and bearing in mind that the cabin load would remain unchanged, a reduction by 50% of the Avionics load would save some 35% of the original 700kW engine penalty.

It should be noted, that for an engine air derived refrigeration system as discussed, then the equivalent engine penalty for cooling a fixed Avionic load of 30kW would be reduced by half if the fixed load was reduced to 15kW.

G German:

The Radar invariably presents the largest heat loads and because of user requirement and the little improvements in thermal efficiency of RF components, it is difficult to see how avionic heat loads can be reduced by significant extent. Is it not possible to make some improvements in the effectiveness of the ECS?

K Morgan

There is no simple answer to this question. If by your question you mean can we improve the effectiveness of the ECS means reducing the amount of energy expended by the engine(s) in cooling the avionic load, then the answer is yes. However to gain significant improvement in the ECS Coefficient of Performance would involve a fundamental re-think on system philosophy leading ultimately to an overall aircraft energy policy involving all systems requiring cooling.

AVIONICS COOLING ON USAF AIRCRAFT

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United States

SUMMARY

There is a continual effort by the United States Air Force (USAF) to provide improved avionics cooling. The purpose of this investigation was to review the methods currently used for cooling avionics equipment on today's USAF aircraft and outline approaches which will improve avionics reliability, reduce aircraft penalty and lower life cycle cost.

This investigation has shown the need for placing greater emphasis on producing compatible avionics equipment and cooling systems. A comprehensive trade study of avionics reliability versus environmental control system (ECS) cooling capability should be conducted at the start of the aircraft development program. From this trade study, the ECS cooling capability can be optimized to minimize total aircraft life cycle cost.

The avionics contractor should conduct a detailed analytical thermal analysis of internal "black box" temperatures early in the development program. During qualification testing, a thermal verification test should be conducted to verify that all component temperatures are within the necessary limits for required reliability.

This study indicates that use of narrow limits on inlet coolant temperature, greater use of cold plates and liquid cooling, and use of ECS approaches similar to the Advanced Environmental Control System should be seriously considered for new aircraft design.

1. INTRODUCTION

One of the prime functions of an aircraft environmental control system is to provide that cooling necessary to maintain internal components of avionics equipment at temperatures which will achieve a long and predictable service life. This can only be achieved by a cooperative effort of both avionics and ECS engineers to produce compatible avionics equipment and cooling systems. Proper maintenance and operation of the ECS in service is also required.

The type of cooling has an important affect on avionics design, performance and reliability. Today's aircraft have a large amount of expensive avionics and the cost of maintaining this equipment represents a large expense to the USAF. Since cooling can have a significant affect on this overall avionics cost, there is a continuing effort by the USAF to provide improved avionics cooling.

The avionics cooling load on today's aircraft significantly influences the overall ECS capacity. In many cases, the avionics load is the major heat load cooled by the ECS. Avionic loads are tending to increase with each new aircraft. Since the ECS imposes penalties on aircraft performance due to the use of ram air and engine bleed air, it is necessary to pursue ECS approaches, which minimize aircraft penalties, in order to counteract the continuing increase in avionics cooling loads.

The purpose of this study was to review the avionics cooling methods used on recent USAF aircraft, and define cooling approaches which improve avionics reliability, reduce aircraft penalty and lower life cycle cost.

2. AVIONICS COOLING DESIGN REQUIREMENTS

The current general design requirements for avionics cooling on USAF aircraft are contained in military specifications MIL-E-38453 and MIL-E-5400 and military standard MIL-STD-454. The environmental control systems for new aircraft are designed to meet the requirements of MIL-E-38453. The avionics equipment for new aircraft are designed in accordance with the requirements of MIL-E-5400. Thermal design requirements for avionics equipment are contained in MIL-STD-454.

The ECS is required by MIL-E-38453 to have the capability to provide avionics cooling in accordance with the detail specifications for the avionics equipment. The ECS must have sufficient cooling capacity to provide cooling for an avionics heat dissipation load 25 percent greater than the on-board load of the first production aircraft. For compartments containing avionics in accordance with MIL-E-5400, the ECS must maintain the compartment ambient temperature within the temperature limits for the particular class of equipment as defined by MIL-E-5400. Cooling air forced directly over the surface of miniaturized or basic electronic components is required to be totally void of entrained moisture. The range and rate of fluctuation between minimum and maximum avionics operating temperature shall be minimized. The inlet cooling air temperature and flow rate to forced and ambient cooled avionics should be controlled to prevent overcooling and assure no problems due to moisture. In addition, MIL-E-38453 requires filtration of air delivered to the interior portions of internally forced convection air cooled avionics.

Graphs are presented in MIL-E-5400 showing the minimum and maximum allowable ambient temperature for avionics equipment versus altitude for four different classes of equipment. In addition, MIL-E-5400 references MIL-STD-454 for thermal design requirements. The requirements of MIL-STD-454 specify that adequate thermal design shall be employed to maintain equipment parts within their permissible operating temperature limits. The simplest and most efficient thermal design shall be selected. Forced air cooling shall be used only when natural cooling does not provide sufficient cooling or when a significant reduction in overall size and weight can be realized. The thermal requirements of MIL-STD-454 require that aircraft

avionics cooled by externally supplied cooling air be designed using cold plates or heat exchangers so that none of the cooling air will come into contact with internal parts, circuitry, or connectors. In addition, MIL-STD-454 states that other cooling methods such as liquid, evaporative coolants, and use of vapor cycle refrigerants shall not be used except when natural or forced air cooling methods are unsuitable.

As can be seen from reviewing the above requirements, the main concerns to be met are to maintain component temperatures within their acceptable minimum and maximum limits and use the most simple cooling approach possible. These kind of design requirements have resulted in many of our ECS being designed with only sufficient cooling capability to maintain avionics components just at or below their maximum allowable temperature under hot day conditions. Many of the systems allow the coolant temperature to fluctuate throughout the complete wide temperature range of MIL-E-5400. The requirements do not force the aircraft contractor to optimize the avionics/ECS interface. Avionics reliability is strongly influenced by the operating temperature of the equipment. The reliability is decreased as the temperature of the components is increased. Military handbook MIL-HDBK-217 shows the reliability versus temperature for many different types of electronic equipment. Avionics reliability can be improved substantially by maintaining the components at temperatures appreciably below the maximum allowable. Studies (HILBERT, W.F., and KUBE, F.H., 1969, and CAWTHON, D.M., and GONZALEZ, J.I., 1969) indicate that avionics reliability can be improved by holding the components at near constant temperature. Use of liquid cooling could result in less overall aircraft penalty; however, the specification requirements tend to negate use of liquid cooled avionics. ECS weight penalty is reduced as coolant supply temperature increases. With liquid coolant, the supply temperature to avionics can be increased over that used with air cooling without an increase in maximum avionics temperature due to the higher specific heat of liquid.

The ECS influences a number of avionics environments which affect avionic reliability. Environments influenced by the ECS and their effects are:

a. Equipment reliability decreases as temperature of the components increases.

b. Temperature cycling reduces equipment reliability.

c. Moisture and dust reduce equipment reliability.

The environments can account for an appreciable number of avionics equipment failures. One study (DANTOWITZ, A., HIRSCHBERGER, G., and PRAVIDLO, D.) showed that over a two-year period, 52% of all failures were environmentally related on thirty-one equipments on approximately 175 aircraft. Of these failures, 21% were caused by temperature, 10% by moisture, and 3% by sand and dust. The substantial number of avionics failures due to temperature, moisture, and sand and dust stress the imporantance of adequate design requirements in these areas in the ECS and avionics specifications.

3. AVIONICS COOLING APPROACHES

A number of different approaches are employed in today's aircraft for cooling avionic equipment. The system cooling approaches and individual equipment cooling approaches most commonly used on USAF aircraft are described in the following subparagraphs.

3.1 SYSTEM COOLING APPROACHES

a. Ram Air - In this approach, ram air is ducted from an air scoop directly to the avionics equipment or bay. Exhaust air may, in some cases, be collected and ducted overboard to prevent excessively high compartment ambient temperatures. Ram air cooling may be used in applications where the maximum ram air temperature is sufficiently low to maintain proper avionics temperatures. Due to the wide variations in available ram air temperature and pressure throughout the flight envelope, the amount of cooling provided will vary from approximately that which is required, to several times the required quantity, for a typical unmodulated ram air cooling system. Ram air cooled avionics should therefore be designed for adequate cooling using the minimum required quantity of airflow, and should be relatively insensitive to large variations in the quantity of airflow, pressure and temperature supplied. When a direct ram air cooling system is used, consideration should be given to the adverse effects caused by entrained moisture in the cooling air. Moisture separators have not been provided in the ram air cooling ducts of many of today's existing aircraft. Entrained moisture can exist in the ram air during flight through clouds and rainstorms and during icing conditions. Moisture problems can also be created during descent from high altitude to a low level flight condition. For this case, warm moist air may condense on cold equipment, resulting in an extremely wet operating condition. During a subsequent climb back to high altitude, freezing of any unevaporated condensed moisture will occur.

b. Air Cycle Discharge Air - In this approach, cooling air is ducted directly from the outlet of the air cycle air conditioning package to the avionics equipment or compartment. Most aircraft have an air cycle air conditioning system. This type of system cools engine compressor bleed air through ram air heat exchangers and expansion through a turbine. Wide variations in avionics cooling air quantity and temperature are possible with an air cycle system, unless constant flow and temperature controls are incorporated. Possible adverse effects due to moisture must be considered for avionics cooled directly by an air cycle system. This is because the moisture separators used in air cycle systems are not 100% efficient in removing entrained moisture from the cooling air.

c. Compartment Exhaust Air - In this approach, occupied compartment (cockpit, passenger compartment, cargo compartment) air is exhausted overboard through the avionics equipment or avionics bay. With this approach, a fairly stable inlet air temperature to the avionics is achieved since the occupied compartment temperature is maintained at a relatively constant level. Adverse effects due to moisture should not result from this approach but possible problems due to dust in the air are created. Dust, which infiltrates the aircraft while doors are open on the ground, can be present in the compartment exhaust or recirculated air. Therefore, filtration is usually necessary for applications where this air is ducted directly to the interior portions of the avionics equipment.

d. Liquid Coolant Loop - In this approach, a liquid coolant is pumped to the avionics equipment and the heat absorbed from the avionics is dissipated into a heat sink such as ram air, air cycle package air, or fuel. Possible adverse effects due to moisture and dust are eliminated with this cooling approach. In addition, constant avionics coolant flow rate and inlet temperature can be more easily provided with this approach.

e. Recirculated Air - In this approach, avionics compartment air is continuously recirculated from the avionics equipment to a heat exchanger where the heat load is dissipated and then back to the avionics equipment. With this approach, a stable inlet air temperature to the avionics can be maintained. Adverse effects due to moisture and dust should not result from this approach.

3.2 INDIVIDUAL EQUIPMENT COOLING APPROACHES

a. Natural cooling which uses the ambient surrounding the equipment as the heat sink. The heat is transferred to the ambient by conduction, free convection and radiation.

b. Forced air cooling by fan mounted in or on equipment case circulating ambient air over internal components.

c. Forced air cooling with aircraft ECS supplying air which is circulated directly over the components within the equipment case.

d. Forced air cooling with aircraft ECS supplying air to an air-to-air heat exchanger in the equipment case, and a blower in the sealed case providing airflow over the components and recirculating through the opposite side of the heat exchanger.

e. Forced cooling with aircraft ECS supplying air to a cold plate upon which the heat producing components are mounted. Heat from the components is conducted to the cold plate.

f. Liquid-to-air heat exchanger in the equipment case, with aircraft ECS supplying air to the air-side and a pump in the case circulating fluid through the liquid-side and over the components to be cooled.

g. Central aircraft mounted liquid-to-air heat exchanger with aircraft ECS supplying air to the air-side and an aircraft mounted pump circulating fluid through the liquid-side and distributed to the liquid cooled avionics equipment.

h. Liquid-to-air heat exchanger in the equipment case, with a fan mounted in the case circulating ambient air through the air-side and a pump in the case circulating fluid through the liquid-side and over the liquid cooled components.

4. AVIONICS LOADS AND COOLANT CHARACTERISTICS

The heat dissipated by avionics/electrical equipment constitutes a major portion of the total ECS heat load on modern USAF fighter and bomber aircraft. It is also a significant portion of the load on cargo aircraft. Table I shows typical avionics/electrical equipment cooling loads on current USAF aircraft. Also shown is the avionics/electrical load as a percentage of the total ECS load.

TABLE I

TYPICAL COOLING LOADS

Aircraft Type	Avionics/Electrical Load - KW	Avionics/Electrical Load / Total ECS Load
Fighter	20-35	75-80%
Bomber	25-125	60-70%
Cargo	10-22	20-30%

The coolant inlet temperature to avionics equipment varies depending on the particular aircraft and the system cooling approach used. Table 2 shows typical coolant inlet and outlet temperatures for the various avionics system cooling approaches.

As can be seen from Table 2, there is a wide variation in the avionics coolant inlet temperatures used on today's USAF aircraft. Many of the current air cycle systems maintain a high inlet temperature at low altitudes in order to be above the dewpoint and prevent moisture problems.

TABLE II

TYPICAL COOLANT TEMPERATURE

System Cooling Approach	Inlet Temperature	Outlet Temperature
Ram Air	-65°F to 120°F (-54°C to 49°C)	160°F (71°C) max
Air Cycle Air		
F-4	Below 25,000 ft. 85°F (29°C) Above 25,000 ft. 40°F (4°C)	145°F (63°C)
F-15	Below 35,000 ft. 83°F (28°C) Above 35,000 ft. 53°F (12°C)	140°F (60°C)
F-16	Above 31,000 ft. & M-1.2 0°F (-18°C) All other conditions 35°F (2°C)	140°F (60°C)
F-111	-65°F to 80°F (-54°C to 27°C)	160°F (71°C) max
Cabin Exhaust Air	70°F to 100°F (21°C to 38°C)	160°F (71°C) max
Liquid Coolant Loop		
F-15	115°F (46°C)	145°F (63°C)
B-1	90°F (32°C)	160°F (71°C) max
Recirculated Air		
B-1	90°F (32°C)	140°F (60°C)

5. AVIONICS COOLING CRITERIA

The following criteria should be considered when selecting the method of cooling to be used for a piece of avionics equipment.

a. Quantity of heat to be dissipated.

b. Watt density of the avionic equipment package.

c. Equipment tolerance to variations in coolant flow rate and inlet pressure and temperature.

d. Equipment tolerance to ambient temperature variations.

e. Equipment tolerance to moisture and dust in the cooling air.

f. Ground and flight conditions during which equipment will be operated.

g. Length of operating time.

h. Explosion proofing requirements.

i. Need for a pressurized ambient for the avionics equipment.

j. Effect of the avionics heat rejection on cabin or equipment compartment ambient temperature.

k. Location of equipment in aircraft.

l. Required coolant flow rate versus inlet temperature.

m. Coolant flow path pressure drop versus flow rate.

6. AVIONICS/ECS COMPATIBILITY

The lessons learned from past and present USAF aircraft programs have shown the need for placing greater emphasis on producing compatible avionics equipment and cooling systems. Avionic equipment reliability is strongly influenced by the operating temperature of the equipment. As a result, it is highly important that proper consideration be given to cooling requirements early in the avionics design period. Such consideration will eliminate the need for many future modifications which are caused by failure to consider all aspects of the available aircraft cooling air supply. A cooperative effort of both avionics and ECS engineers is required in order to assure compatibility between the ECS and avionic equipment. In order to enhance compatibility, the following actions need to be taken:

a. At the start of the aircraft development program, the airframe contractor should conduct a comprehensive trade study of avionics reliability versus ECS cooling capability. The trade study should consider all factors affecting life cycle cost and aircraft penalties. The study would consist of determining the overall avionics reliability for various levels of avionics cooling. The levels to be considered would be a wide range of coolant inlet temperatures such as 0°F to 90°F (-18°C to 32°C). For each of the inlet temperatures studied, a range of exit temperatures such as 100°F to 160°F (38°C to 71°C) should also be

considered. For each of the inlet/exit temperature combinations studied, the effect on ECS weight, volume and development cost, and effect on aircraft penalties should be determined. The overall effect on life cycle cost should then be determined for each of the combinations and the ECS/Avionics Cooling approach, resulting in minimum life cycle cost should be selected. In order to conduct a meaningful trade study early in an aircraft development program, it is necessary to have a fairly adequate definition of the avionics package and it is necessary to be able to predict the reliability of the avionics as a function of temperature. Reports (CAMPBELL, S.A., and TAYLOR, K.J., 1971) are available which give information on conducting the trade study of avionics reliability versus ECS cooling capacity.

b. Each avionics contractor should conduct a detailed analytical thermal analysis on each "black box" early in the design phase. The analysis should predict internal component temperatures for various operational conditions. This is necessary for justifying that the amount of coolant flow stated as necessary will maintain internal component temperatures at or below the level necessary to achieve the required reliability. Transient thermal analyzer computer programs (KOCYBA, T.E., 1974) are available for conducting the thermal analysis.

c. The avionics contractor should conduct a thermal verification test on each piece of avionics. The test should be conducted on prototype or pre-production equipment, representative of the design to be released for production. The avionics equipment should be operated at conditions of maximum heat dissipation in the worst case specified environment with cooling equivalent to that provided on the aircraft. The equipment should be instrumented so that temperatures are monitored (1) for all critical parts, (2) for any part whose individual dissipation is 1% or more of the total equipment dissipation, and (3) such that the sum of the dissipation of the monitored parts is 90% or more of the total unit heat dissipation. The purpose of this test is to verify that the cooling provided and the internal arrangement of components within the "black box" will result in component temperatures which are at or below the levels necessary to achieve the required reliability.

d. Provisions should be incorporated into the aircraft which prevent operation of the avionics equipment on the ground unless adequate cooling is provided. Experience has shown that many times avionics are operated on the ground during maintenance without providing any cooling to the equipment. This leads to reduced avionics reliability. One approach for preventing this occurrence on future aircraft is to incorporate an interlock which prevents providing electrical power to the avionics equipment unless adequate coolant is being delivered to the avioncs.

7. ADVANCED ENVIRONMENTAL CONTROL SYSTEM

The Air Force Flight Dynamics Laboratory at Wright-Patterson AFB is currently sponsoring development of an Advanced Environmental Control System (AECS). One of the main objectives of this program is to develop an ECS which will provide a stabilized environment for the avionics. The AECS is designed to provide a constant supply of clean, dry 40°F (4°C) air to the air cooled avionics. The cool inlet air, which is free of entrained moisture, and the constant inlet temperature, should result in improved avionics reliability.

The AECS is being developed under a three year contract, which began in May 1974, with McDonnell Aircraft Company (MCAIR) as the prime contractor and Hamilton Standard as a major subcontractor. The system is being designed for flight test on an F-15 test aircraft. MCAIR will complete breadboard and brassboard laboratory testing of the AECS by the end of 1976. Flight testing of the system is scheduled to start in May 1977.

Figure 1 is a schematic of the AECS system. It is designed to operate on minimum engine idle bleed pressure of 35 psia. The system is designed to deliver 40°F (4°C) cooling air having no more than 25 grains of water per pound of dry air when operating in an ambient condition of 105°F (47°C) dry bulb, 200 grains of water per pound of dry air. The following innovations are in the AECS:

a. Fuel Heat Sink & Wing Tank Fuel Heat Rejection - A portion of the secondary heat exchanger heat load is rejected to fuel via a Coolanol 25 heat transport loop. Fuel is sprayed into the wing fuel tanks when fuel flow for cooling exceeds that required by the engines. Heat absorbed by the fuel which is sprayed into the wing tanks is then rejected to ambient through the wing skin.

b. Rotating High Pressure Water Separator - Water is condensed on the high pressure side of a rotating crossflow heat exchanger. Centrifugal forces separate the water and it is removed from the delivery air. Cold turbine discharge air flows through the low pressure side and serves as heat sink for condensation on the high pressure side.

c. Variable Geometry Air Cycle Machine - The turbine nozzle area varies as a function of bleed air pressure and cooling load flow requirements. The compressor exit diffuser vane angle also varies as a function of turbine nozzle position to maintain compressor efficiency and extend its surge margin.

d. Water Depressed Regeneration - Water collected from the water separator is sprayed into the cabin and avionics discharge air to lower its temperature. The cabin and avionics discharge air then flows through a regenerative heat exchanger, absorbs a part of the secondary heat exchanger heat load and is then exhausted overboard.

The major constituents of the Advanced Environmental Control System are the bleed air system and the air cycle air conditioning system (ACACS). The bleed air system preconditions the engine bleed air to a lower temperature and a regulated pressure. The ACACS further conditions the bleed air to achieve the desired delivery temperature, remove the moisture from the air and regulate the AECS air flow rate to that required for crew station and avionics. The ACACS controls the avionics delivery temperature by controlling the secondary heat sink. The ACACS controls the AECS air flow rate by modulating the air cycle machine nozzle area. The bleed air system and ACACS are described in detail as follows:

a. Bleed Air System - Bleed air is extracted from each engine at the 13th compressor stage, ducted through the primary pressure regulator and shutoff valve set at 75 ± 15 psig and then to the backup bleed air pressure regulator and shutoff valve set at 120 ± 20 psig. The air is then routed to the primary heat exchanger and to the primary heat exchanger bypass valve. At ram air temperatures above 80 ± 5°F, all bleed air is routed through the primary heat exchanger allowing the bleed air to be cooled to 150°F or the maximum capacity of the primary heat exchanger whichever is higher. At total temperatures below 80°F, the bleed air temperature sensor will control the air flow through the heat exchanger by modulating the bypass valve to maintain a preconditioned bleed air temperature of 325 ± 20°F. This temperature level control provides the means to achieve cabin and anti-fog heating and maintain the ACACS compressor component operating conditions out of surge. This control will also provide 325°F when the anti-ice system is selected. Protection from excessive temperature consistent with auto-ignition temperature requirements is provided by the bleed air over-temperature sensor which pneumatically modulates the backup bleed air pressure regulator/shutoff valve to the closed position during over temperature conditions. The primary heat exchanger utilizes ram air as a heat sink to provide initial conditioning bleed air. At air speeds above 0.80 Mach, ram air is extracted from the engine air inlet ducts. At air speeds below 0.80 Mach, an auxiliary ram air door extends into the airstream, to provide the required ram airflow for bleed air cooling. Check valves, located in the primary heat exchanger ram air inlet duct downstream of the engine plenums, prevent reverse flow into the engine inlet ducts. On the ground and at air speeds below 180 knots, the primary heat exchanger ejector valve opens on a signal from the Air Data Computer to allow engine bleed air to be ejected in the primary heat exchanger ram air exhaust duct. The induced airflow through the primary heat exchanger is sufficient to provide bleed air cooling during ground operation. Check valves, located downstream of each primary heat exchanger, prevent bleed air crossflow and permits independent operation of each engine bleed air system. Air leaving both primaries is ducted together and then flows through a dust separator where 75% of the dust is removed.

b. Air Cycle Air Conditioning System - Preconditioned engine bleed air enters the air cycle air conditioning system through the turbine/compressor inlet duct. The airflow is ducted to the turbine/compressor regulator and shutoff valve which controls at 60 ± 6 psig and to an auxiliary air duct which provides preconditioned bleed air for fuel system pressurization, windshield anti-ice, canopy seal pressurization and avionics waveguide pressurization. Preconditioned bleed air is compressed by the compressor component of the variable geometry turbine compressor. This compressed air is then routed to the secondary heat exchanger where the heat of compression is transferred to the fuel cooled Coolanol loop. The airflow is then routed through a regenerative high pressure water separator (HPWS) heat exchanger which is designed to cool the air below the saturation temperature equivalent to 25 grains of water per pound of dry air by means of the turbine discharge air. The air is then routed to the turbine/compressor where it is expanded by the turbine component. Upon discharge from the turbine the air is routed through the cooling side of the HPWS. The Cold Side discharge air from the HPWS is controlled to $40 ^{+5}_{-0}$°F by controlling the quantity of heat removed in the secondary heat exchanger. The Coolanol loop heat load is ultimately transmitted to the fuel and the avionics and cabin discharge air. Downstream of the HPWS the airflow is divided, with one branch to the cabin conditioning system and one branch to the avionics cooling air distribution system. Sensing the avionics flow and extracting the cold air for the crew station upstream of the avionics flow sensor allows the ACACS to achieve the minimum required flow rate. The basic AECS flow control is achieved by modulating the turbine nozzle area in response to the avionic flow sensor. The avionics flow rate is set at 39.3 lb/min for weight on wheels (ground operation) and 51.8 lb/min for flight operation.

The AECS is designed to provide improved cockpit and avionics cooling and at the same time, reduce aircraft penalties. The variable geometry air cycle machine and the lower supply air temperature to the avionics results in less bleed air consumption. The use of fuel and cabin/avionics exhaust air as a heat sink results in less ram air drag because a ram air cooled secondary heat exchanger is not required. The elimination of the ram air cooled secondary heat exchanger also results in reduced system volume.

8. CONCLUSION

The importance and complexity of avionics cooling cannot be overemphasized. As a result of this investigation, the following should be done in order to improve avionics reliability on future USAF aircraft:

a. The avionics contractor should conduct a detailed analytical thermal analysis of internal component temperatures early in the development program.

b. During qualification testing, a thermal verification test should be conducted to verify that all component temperatures are within the necessary limits for required reliability.

c. Coolant inlet temperature and flow rate to avionics equipment should be maintained within narrow limits.

d. All new avionics equipment should be designed for "cold plate" cooling.

e. Provisions should be incorporated into the aircraft design to prevent operation of the avionics on the ground unless adequate cooling is provided.

In order to reduce the resulting aircraft penalty due to avionics cooling on future USAF aircraft, the following should be done:

a. ECS approaches similar to the Advanced Environmental Control System should be seriously considered. Approaches should be stressed which maximize use of fuel as a heat sink and minimize use of ram air and engine bleed air.

b. Wider use should be made of liquid cooled avionics equipment.

In order to reduce aircraft life cycle cost, a comprehensive trade study of avionics reliability versus ECS cooling capability should be conducted at the start of the aircraft development program. This will then allow selection of the ECS/Avionics Cooling approach resulting in minimum life cycle cost.

This investigation has also shown that current U.S. military specifications for avionics equipment and environmental control systems need to be revised to have compatible and adequate requirements for temperature, pressure, moisture and dust control. The avionics specifications need to include a firm requirement for a thermal analysis and thermal verification test.

9. REFERENCES

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AECS Schematic

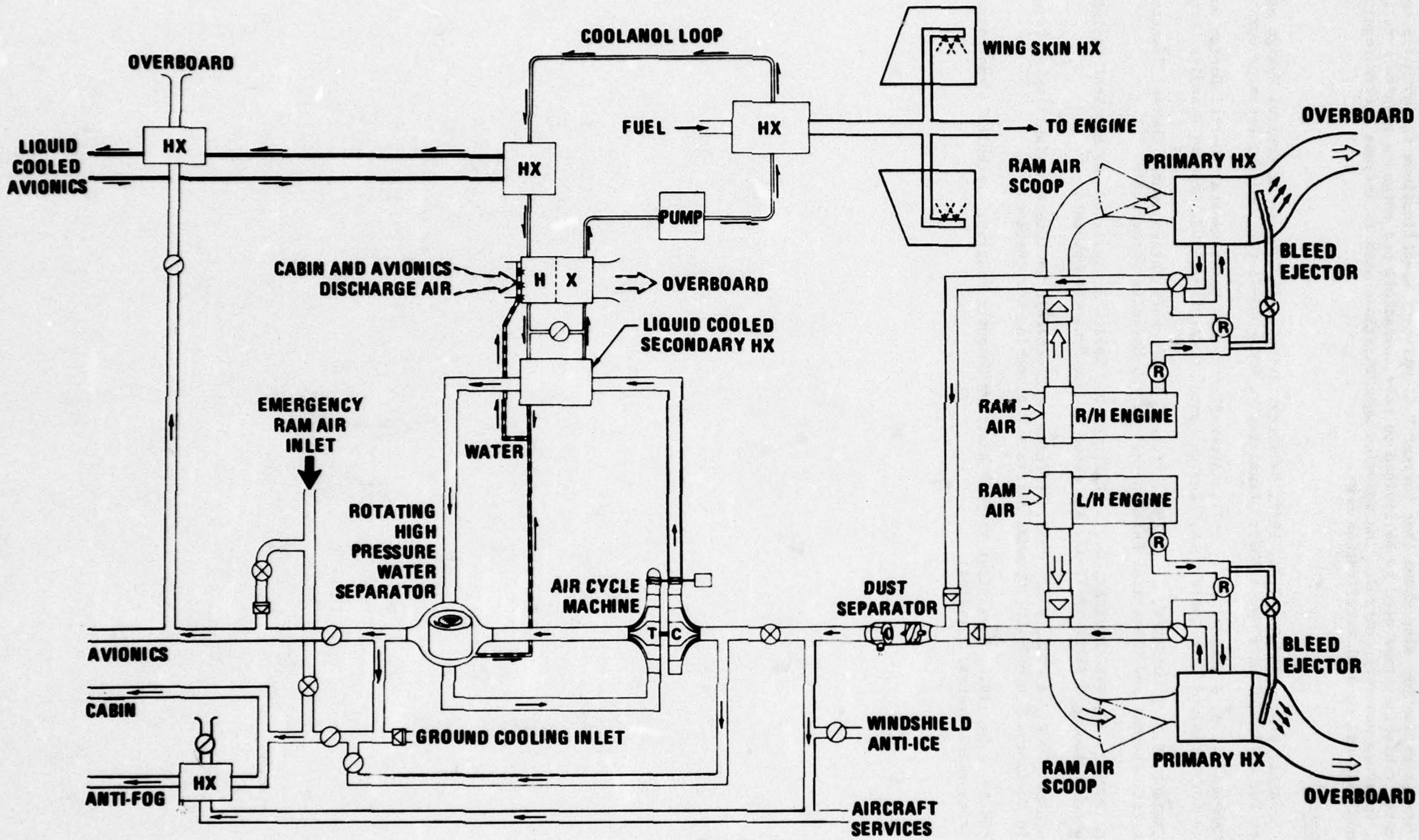

Figure 1

DISCUSSION

J Wheeler:

My question is in two parts. Firstly, is 25 grammes of water per pound of air a dew point of 40°F?

Secondly, would the Author think that this was low enough for an EHT electrical problem in a radar equipment?

G C Letton Junior:

The answer to the first part of your question is no, 25°F (approximately).

As for the second part, I have little knowledge of radar as a subject; however, this level is not thought to be low enough for the particular problem. Radar is normally cooled by a separate cooling system which employs a dessicator to ensure that the air is dry enough.

G R Giles:

Are there any quantitative values for the reduction in aircraft penalty for the F-15 AWCS compared with the conventional F-15 AWCS?

G C Letton Junior:

There are no figures available at present, they will be available when the Report is issued by Macdonald Aircraft as an Airforce Flight Dynamics Technical Report.

G W Underwood:

(i) Is the exercise intended as an improvement to an existing F-15 system or is it a research exercise on optimising an EC system?

(ii) Is fuel used as the main cooler in the AECS and if not can you tell me more about the system?

G C Letton Junior:

(i) The Advanced Environmental Control System (AECS) is not intended as an exercise to improve the current F-15 system. The current F-15 system is considered to be completely adequate. The purpose of the AECS is to demonstrate an advanced system approach and to look at components which will improve Avionics Cooling and reduce aircraft penalties.

(ii) No, fuel is not used as the main heat sink for AECS. Ram-air, and cabin and avionics exhaust air are used as heat sinks. These are accompanied by water removal. Cabin and avionics exhaust air are an important coolant and have to be made to meet the ground hot day operational design condition.

D Bosman:

Presently, aircraft subsystems (Avionics, Hydraulics, Electrics etc) are designed as closed sub-systems. The aircraft designers task is then to make such independant entities cooperate on a federative basis under some management scheme (firmware, software, human factors, and etc).

Does your study also show that shifting away from the federative concept towards integration could markedly improve cooling efficiency?

G C Letton Junior:

My Study only addresses the integration of Avionic Cooling requirements with environmental control system cooling capacity. This integration will improve cooling efficiency. However, there would also be a pay-off in further integration such as the integration of secondary power and ECS and Integrated Thermal Management.

ADVANCED THERMAL COMPONENTS FOR EFFICIENT COOLING OF AVIONIC SYSTEMS

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SUMMARY

A brief description of avionics cooling requirements is given and the overall design features of Spacelab avionics cooling system are presented.The analytical tools for comprehensive calculations in the area of cooling systems are discribed. Different types of heat pipes and phase change components are presented showing the possible kinds of applications. These heat transport and storage components mainly were developed for space application. The successful results encourage us to utilize these components also in similar fields as avionics cooling.

Introduction

One outstanding feature in the development of military aircraft is the continued increase in the quantity of avionics equipment installed aboard. A further remarkable feature in the aircraft development is the high speed at low altitude which makes the usual cooling concepts (airframe, fuel) less effective.

For advanced military aircrafts both features are serious constraints at present and there is the urgent need for more efficient cooling of avionics.

This paper shall contribute to the selection and innovation of more suitable cooling for avionics equipment by presenting devices and systems applied and developed in space flight.

Avionics Cooling Requirements

The main purpose of the cooling is to keep the avionics temperature within a certain range, which is dependant on the operational behavior of the equipment. Normally, equipment likes it cool resulting in a long Mean Time Between Failure (MTBF). On the other hand equipment shall be kept above a certain temperature to allow turn-on without any warm-up time. Average operation temperature is in the order of $330^{o}K \pm 20^{o}K$.

But there are extremes, requiring cryogenic temperatures, liquid nitrogen (78 ^{o}K) or even less. Beyond the primary requirement to keep the temperature, the cooling system and the coolant itself have to meet much more secondary requirements.

To summarize only some of them, these are

- compatibility with avionics parts
- non–toxicity
- allowance of Line Replaceable Unit concept
- non disturbing noise source
- cost-effectiveness over life of craft
- flexibility due to design changes on avionics side
- overload capacity and growth capability.

Typical Cooling Concepts

The heat, generated by avionics parts may either be transferred to the ambient directly or via the case. In both cases the fundamental heat transfer modes apply:

- radiation
- conduction
- free convection
- forced convection
- evaporation
- phase change

From the variety of cooling systems only the most typical shall be mentioned. The open (air) loop simplest system is to blow over or draw through the equipment air from the cabin. It is an available and reliable system, but it is yielding increased equipment temperature on ground or in low level flight.

The closed air circuit compared to the open loop offers only small advantages in terms of better air quality on the expense of higher mass, power consumption and cost. The heat sink for the air may be a heat exchanger with air on the cold side. A better temperature control is achieved by using the evaporator of a refrigerator cycle instead of the air heat exchanger. But this additional unit increases the penaltys on mass, cost and reliability.

Instead of cooling the equipment with air, it can be placed on mounting plates which are internally cooled by a liquid (coldplates). The liquid then rejects the absorbed heat via a heat exchanger.

Another modification of this concept is to use the mounting plates as evaporators in a refrigerator cycle. This gives an efficient and compact cooling but it is pretty disadvantageous with regard to mass, safety, complexity and cost.

Avionics Cooling Aboard Spacelab

Spacelab is the first step of Europe into manned spaceflight. As a working platform in a near-earth orbit, it provides standardized racks housing electronic equipment (Fig. 1). This equipment will be cooled by air as shown in Fig. 2. Cool air is blown into the rack at the bottom.

Surface cooled equipment will be cooled by air flowing across the surface. The heat transfer is dependent on the actual size of the equipment and the local air velocity. In some specific cases it is necessary to provide some air guide panels around this equipment and suck the air from this area.

High heat dissipating equipment will be cooled applying the suck-thru concept. This means, the box is connected to the rack air return duct, thus air will be sucked from the rack interior through the box, picking up the heat.

Two types of racks will be used. The single experiment rack has only one return duct and one inlet. The double experiment rack, which is twice as broad as the single rack has two inlets and two outlets. The dimensions are roughly 2.77 m height, 0.5/1 m width and 0.8 m depth.

The racks are designed for a total load of 1600 W and 3130 W respectively. The air flow is derived from the ARINC flow equivalent of 0.218 kg/Wh. (ARINC = required airflow to cool a certain amount of electronic). Across the total rack roughly a pressure drop of 5 mbar is required to maintain the design flow. Included in this number are 2.5 mbar for the equipment itself. The air inlet temperature is about 22–24^{o}C nominally. According ARINC flow, the exit temperature is about 40^{o}C. Each rack inlet and outlet is equipped with a manual shut-off valve providing time sharing between various racks during on-orbit. The flow per box or stub is controlled by an orifice.

Thermal Analysis

We performe comprehensive design calculations for general thermal problems and also for the special case of fluid loop cooling using the DORNIER Temperature Field Analyzer (TEFA). This is a numerical differencing analyzer which is designed as a general thermal analyzer accepting resistor-capacitor network representations of thermal systems. The analyzers as par example SINDA, MITAS, LOHARP which have been developed in the USA work on the same basis of lumped parameter representation. This means finite isothermal volumes the so called "nodes" are interconnected to each other by radiation, conduction and convection heat exchange. Various peculiarities may be added e.g. subroutines for pressure drop in tube flow or state changes of gases or vapor including effects of latent heat.

Due to the fact of high flexibility and adaptibility to practically all kinds of thermal design problems these thermal analyzers are a powerful tool also for fluid cooling loops.

Advanced Cooling Components

Heat Pipes

The heat pipe is an efficient thermal energy transfer device. Heat which is imposed on the heat pipe evaporator, vaporizes the working fluid. The vapor moves to any cooler point of the heat pipe and condenses. By this condensation the heat is given to the condenser wall. The fluid then returns to the evaporator section by means of the capillary or gravity forces. Due to this evaporation and condensation cycle the temperature gradient is very low (2 – 5°C dependant on working fluid) (Fig. 3). The operating temperature of the heat pipes depending on the working fluid filled in is for nominal avionics application between –100 and +100°C. Higher and lower temperatures are possible. Typical applications of heat pipes are:

– fixed conductance heat pipes for heat flux transformation
– fixed conductance heat pipe for isothermalizing effects
– heat pipe diode
– gascontrolled heat pipes for stabilizing component temperature level (active controlled ± 0.5°C stability).

Fixed Conductance Heat Pipes

Fixed conductance heat pipes constructed with axial grooved heat pipe profiles (Fig. 4) have been developed for several applications at Dornier System. This heat pipes have the following advantages:

– reproducible capillary structure
– reproducible performance
– low radial and axial temperature gradients
– low cost (material and construction)
– high reliability
– bendable without problems

Fig. 5 and 6 shows some bended fixed conductance heat pipes. The application of these heat pipes to a cold plate is shown in Fig. 7. Heat Pipes imbedded in a honeycomb structure are connected to a fluid cooling channel. The dissipation heat of the electronic components is transferred to the heat pipes and then transported by the heat pipe to the cooling channel and transferred to the cooling fluid.

Heat Pipe Diode

Due to gravity effects of the working fluid a diode operation of the heat pipe is possible. Heat can be transported through the heat pipe only in that direction, which leads to a gravity assisted reflux of the working fluid.

Gas Controlled Variable, Conductance Heat Pipes

Constant temperature or variable conductance heat pipes are designed to provide very efficient temperature control under varying environmental parameters as variable heat load and sink temperatures.

The thermal conductance is automatically varied by changing the active condenser length of the heat pipe. This blocking is done by a non condensible gas which is positioned in a reservoir and parts of the condenser. Temperature control results in the change of vapor pressure in the heat pipe. The non condensible gas controlles the vapor pressure and so automatically the temperature.

At Dornier such heat pipes with integrated radiator have been developed for satellite application. Fig. 8 shows a radiator with 6 integrated gas controlled heat pipes. The control temperature range for this radiator is 30 to 40°C for a heat load change from 40–180 W.

Phase Change Thermal Capacitors

Phase Change Thermal Capacitors are suitable avionics cooling components, able to store the dissipation heat of avionic equipment at constant temperature. They work with a solid/liquid phase change of substances having fusion enthalpies of more than 200 J/g and optional melting points between –100 and +100°C.

Because of the reversibility of the phase change a capacitor, loaded with high power during a short time, could be discharged within a much longer period with low power.

During melting, the local temperatures would remain constant, thereby minimizing the amount of any high temperature excursion. A low temperature fusion material, normally liquid, could also be used to minimize low temperature excursion by release of its heat of fusion when freezing. A typical temperature course of a phase change capacitor is shown by Fig. 9.

Normally a certain heat conduction matrix designed par example as honeycomb material guarantees the heat distribution within the interior of the unit.

Usual phase change heat storage materials are paraffins and their derivates. The substances are compatible with structure materials, non-toxic and suitable for long duration duty. The storage capacity is optional, capacitors up to 400 Wh have been developed. Storage periods of several hours are attainable. For simple joint conductance arrangements heat fluxes of 2000 W/m^2 has been measured. This value can be essentially increased applying forced convective heat exchange. The specific storage capacities are 50–70 Wh/kg for the storage material and approximately 30–50 Wh/kg for the total capacitor.

One type of greater capacitor platforms equipped with various heat dissipating components represents Fig. 10.

The thermal capacitors have been designed according to the following main criteria:

- light weight and compact design
- vacuum tightness of the storage cell
- standardisation of types
- high reliability for long time functioning

Standard capacitor housings are shown in Fig. 13

Possible and typical applications are:

- survival of temperature sensitive equipment during temperature extrema
- thermal stabilization of components (i.e. sensors) within a small allowed range of temperature (± 2°C)
- temperature control of electronic equipment with variable heat dissipation
- constant temperature heat sinks
- temperature peak dumping within cooling fluid loops at variable power profiles
- temperature control during short-term transition conditions of either increased equipment power dissipation or decreased aircraft cooling

As an example, the application of phase change thermal capacitors within the cooling loops of the Space Laboratory should be mentioned here. There is a need of thermal dumping elements twice, first in the water cooling loop, second in the experiment cooling loop, where thermal capacitors are combined with cold plates. This thermal concept has been initiated by Dornier System.

Fig. 11 represents a scheme of the Spacelab Module Cooling Loop. The dissipated peak heat load of 12.4 kW, exceeding the normal heat load of 7,4 kW, has to be stored temporarily by the capacitors.

In the experiment loop, dissipation heat peaks coming from the experiments are absorbed by the capacitors. This results in an equalized temperature course within the cold plates, increasing so the efficiency of the loop.

Being engaged in the field of thermal capacitors for several years, Dornier System has developed and qualified a series of different PC-capacitor types including flight units, represented by Fig. 12. These capacitors have been destined to be integrated as cooling devices into the equipment either by means of conductive heat contact or convective heat transfer within fluid loops.

The first integration type is represented by Fig. 14. The shown thermal capacitor mounting panel is able to absorb peak load coming from the electronic components, so dumping temperature excursions. A PC-heat capacitor combined with a fluid loop is shown by Fig. 15. Here, periodical temperature peaks or sinks can be levelled resulting in a equalized mean fluid temperature.

Fig. 16 shows a Phase Change Constant Temperature Heat Sink. The capacitor is combined with a fluid loop for charging or discharging of the heat sink.

Another application is represented in Fig. 17. Here, the PC-thermal capacitor plate is combined with high heat dissipating electronic equipment, i.e. shunts and transistors.

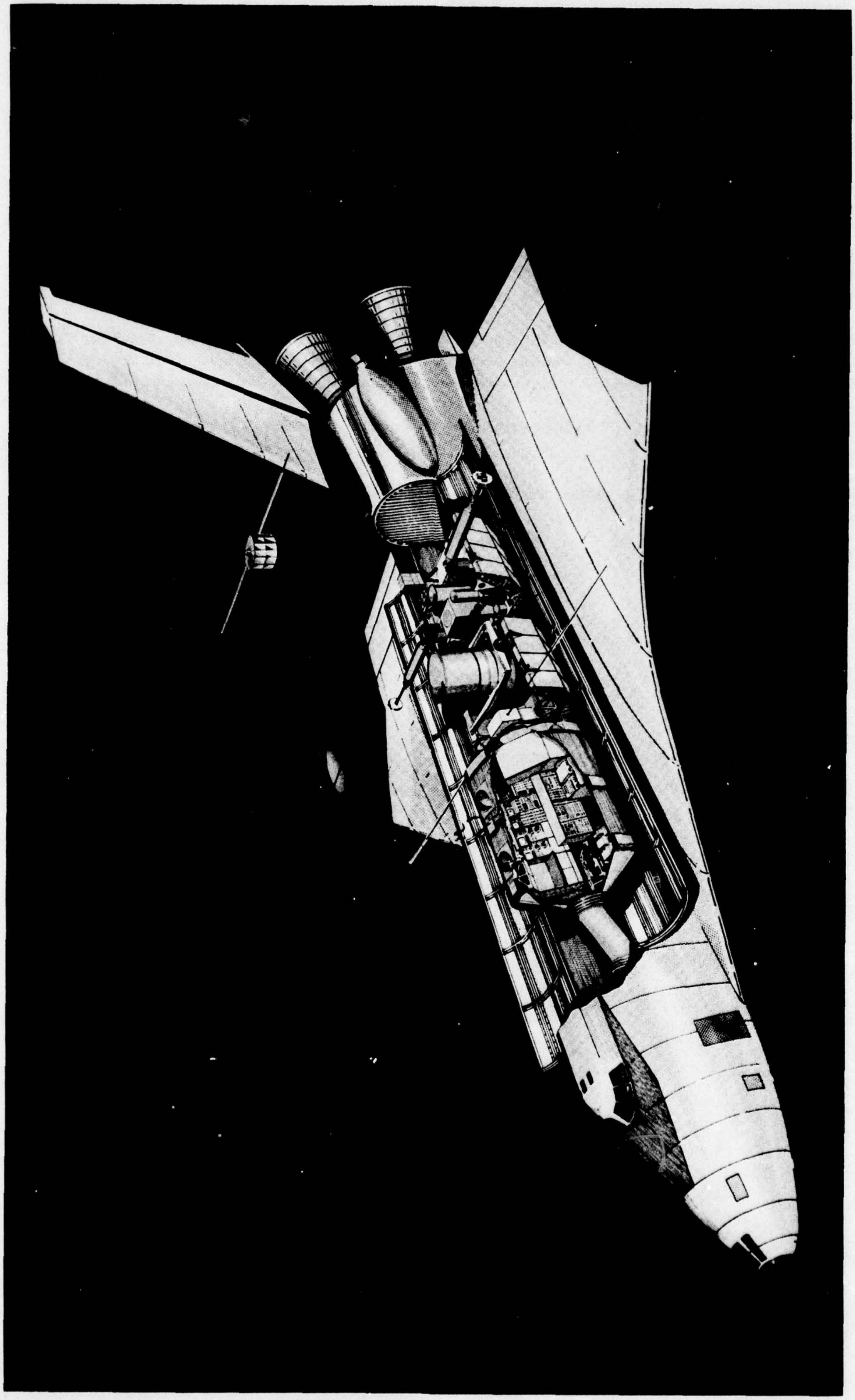


Fig.1 Total view of spacelab

- PROVIDES ADEQUATE AIR FLOW FOR AVIONICS
- SUCTION CONCEPT (COLD FREE AIR STREAM, HOT DUCTED AIR)
- SUCK–THRU COOLING NORMALLY, BUT SURFACE COOLING POSSIBLE
- MASS FLOW PER PORT ADJUSTABLE BY ORIFICES
- IN–FLIGHT SHUT–ON/OFF CAPABILITY OF AIR FLOW TO RACKS
- FOUR DIFFERENT RACK TYPES:
  - WORKBENCH RACK (WBR, COLDPLATED HEAT LOADS)
  - CONTROL CENTER RACK (CCR, AIR COOLING 1056 W)
  - SINGLE EXPERIMENT RACK (SER, 1600 W)
  - DOUBLE EXPERIMENT RACK (DER, 3130 W)
- OPEN POSITION OF SHUT–OFF VALVES VARIED FOR INTER–RACK FLOW BALANCE

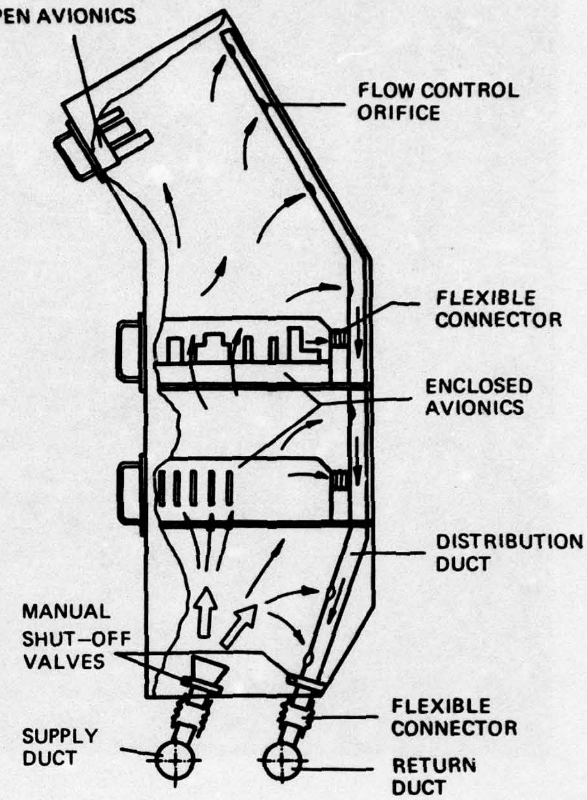


Fig.2 Rack cooling assemblies

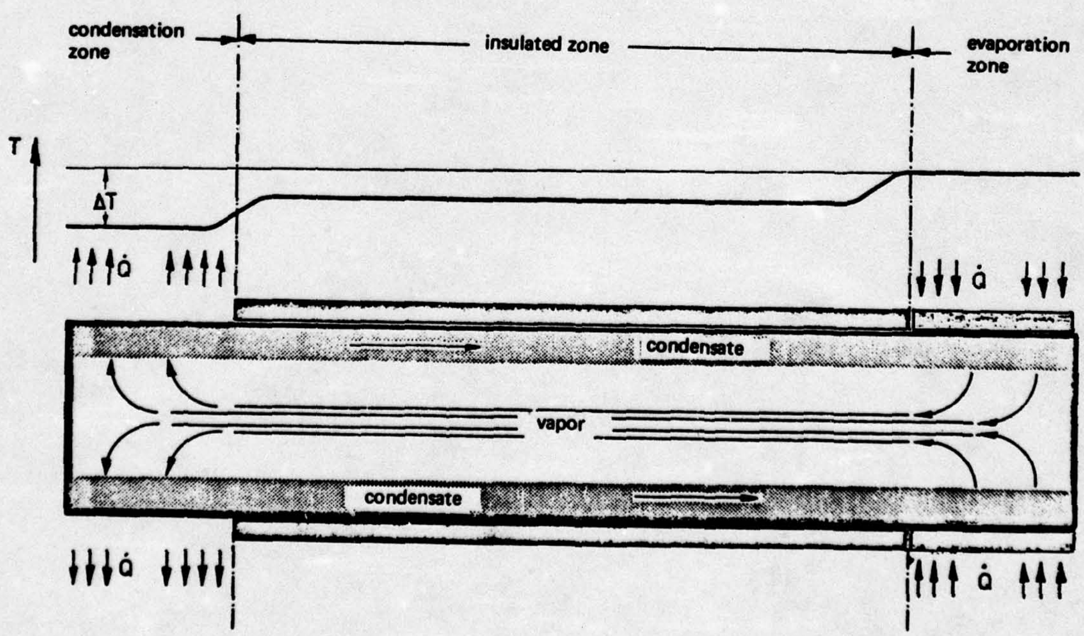


Fig.3 Schematic heat pipe function

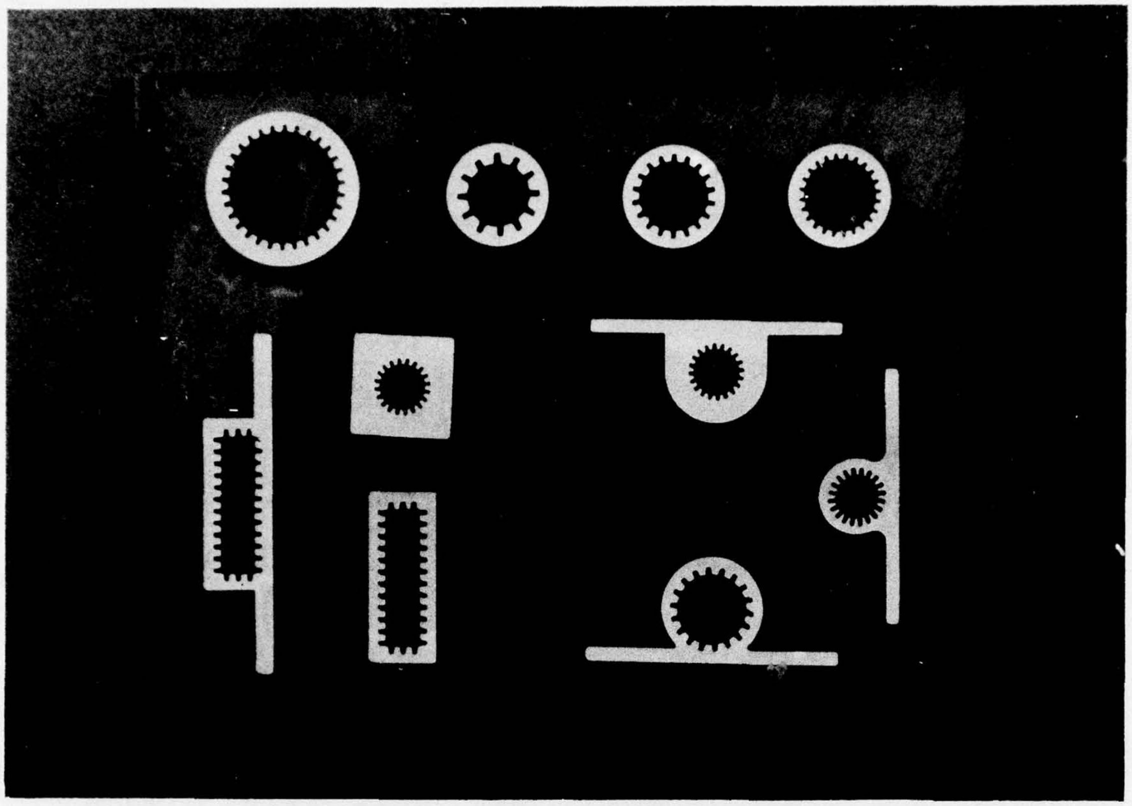


Fig.4 Axial grooved heat pipe profiles

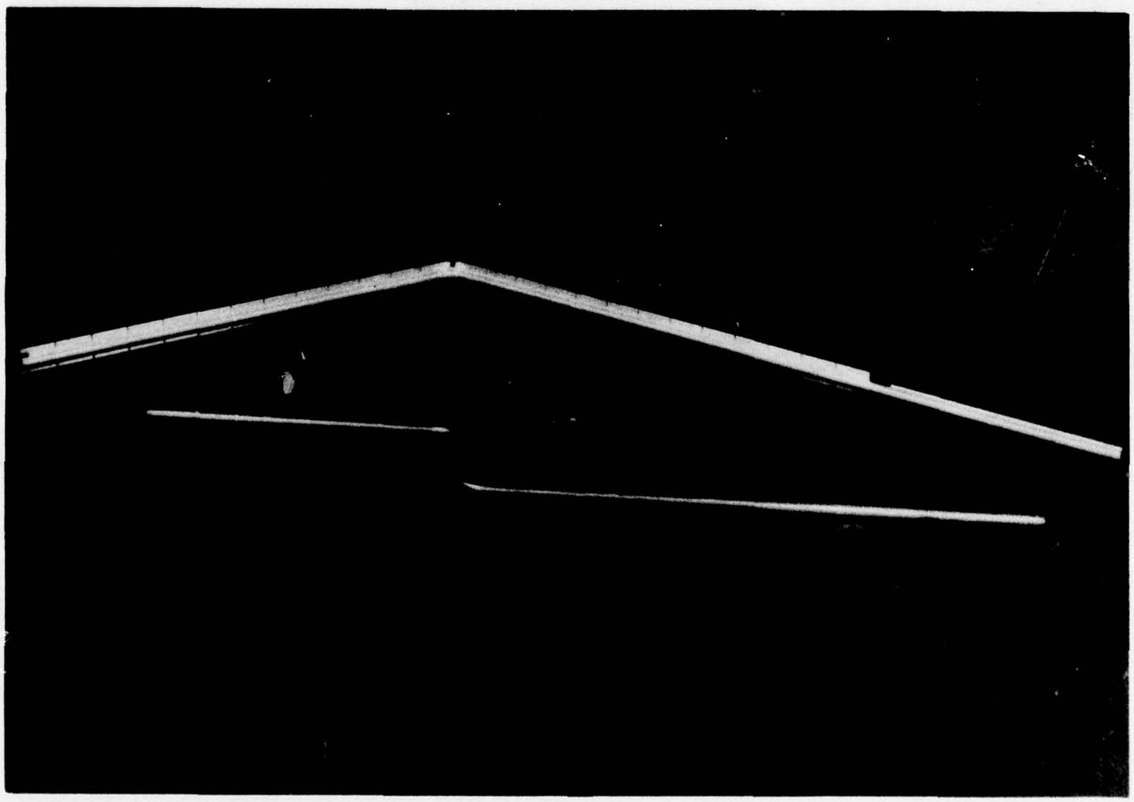


Fig.5 Fixed conductance heat pipes

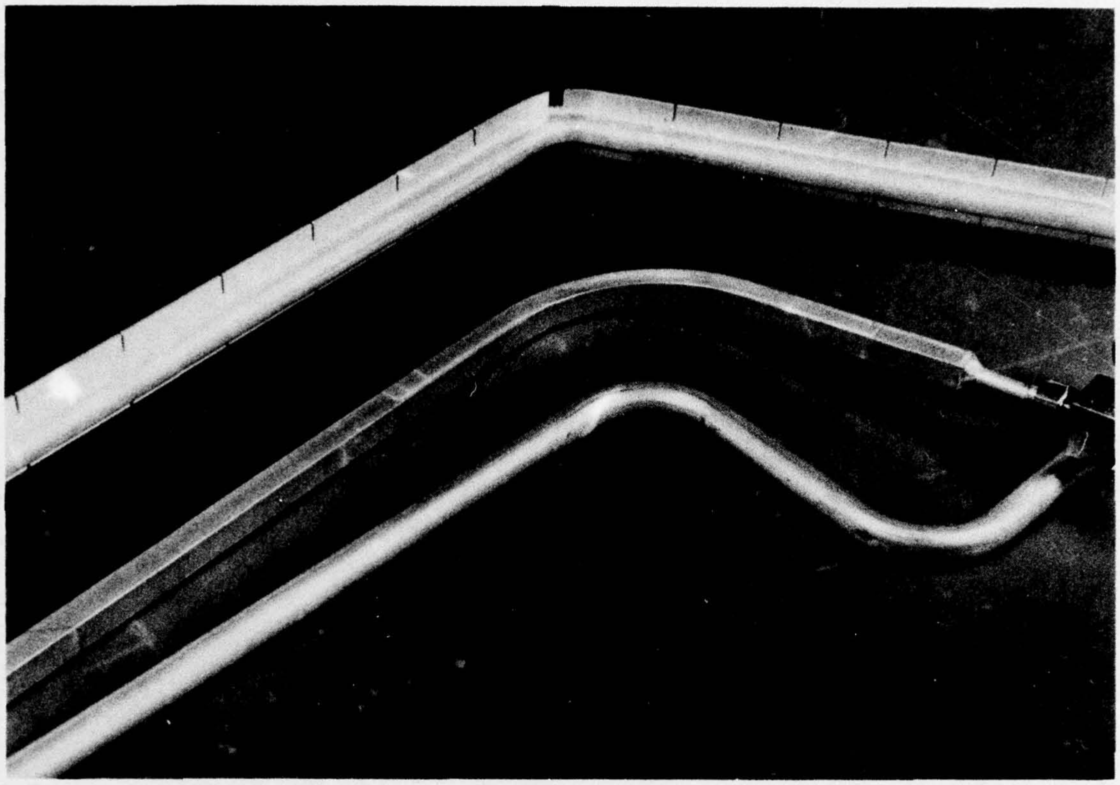


Fig.6 Fixed conductance heat pipes

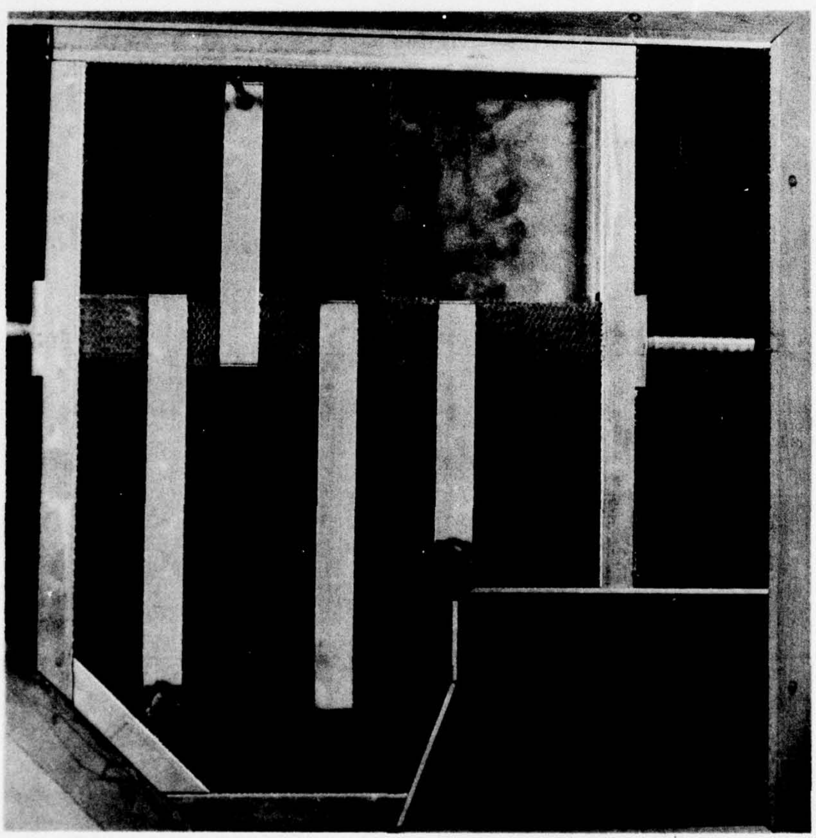


Fig.7 Cold plate with heat pipes and honeycomb structure

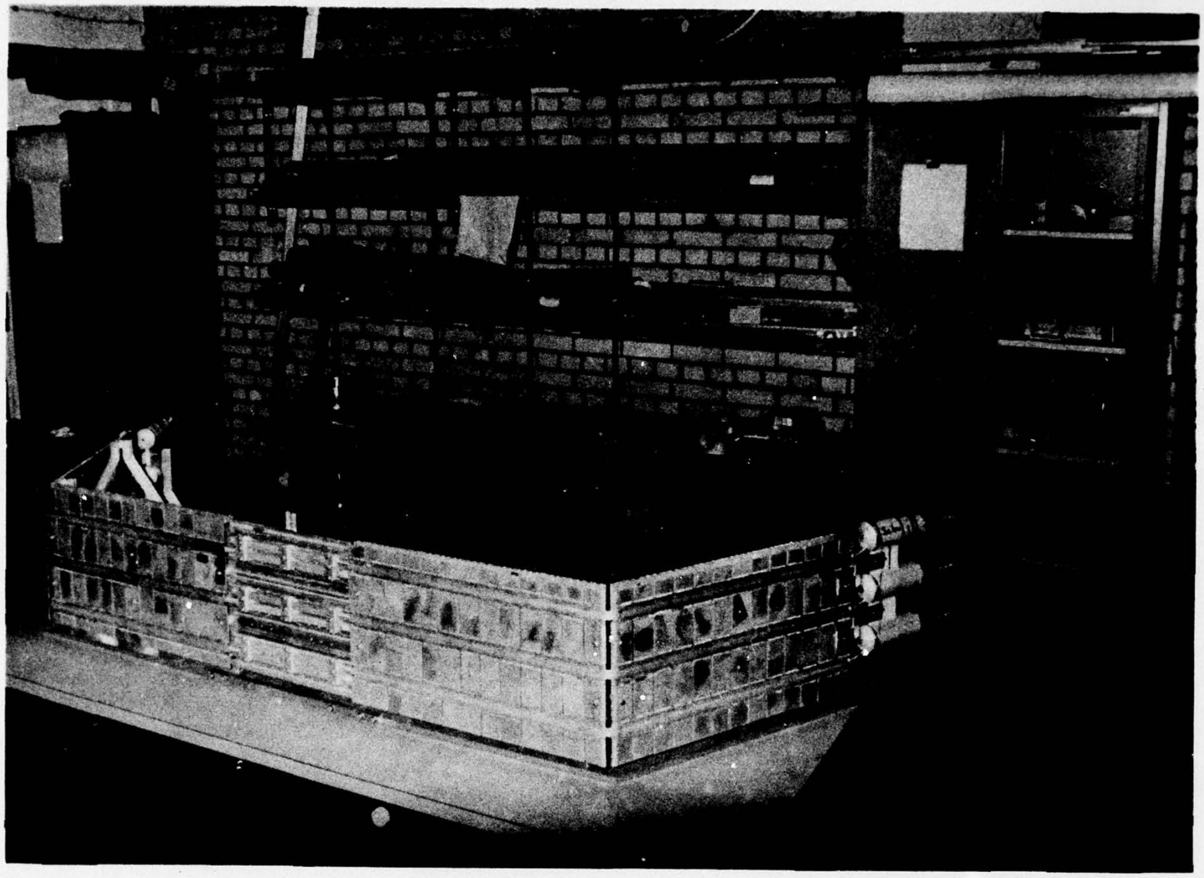


Fig.8 Gas controlled heat pipe radiator

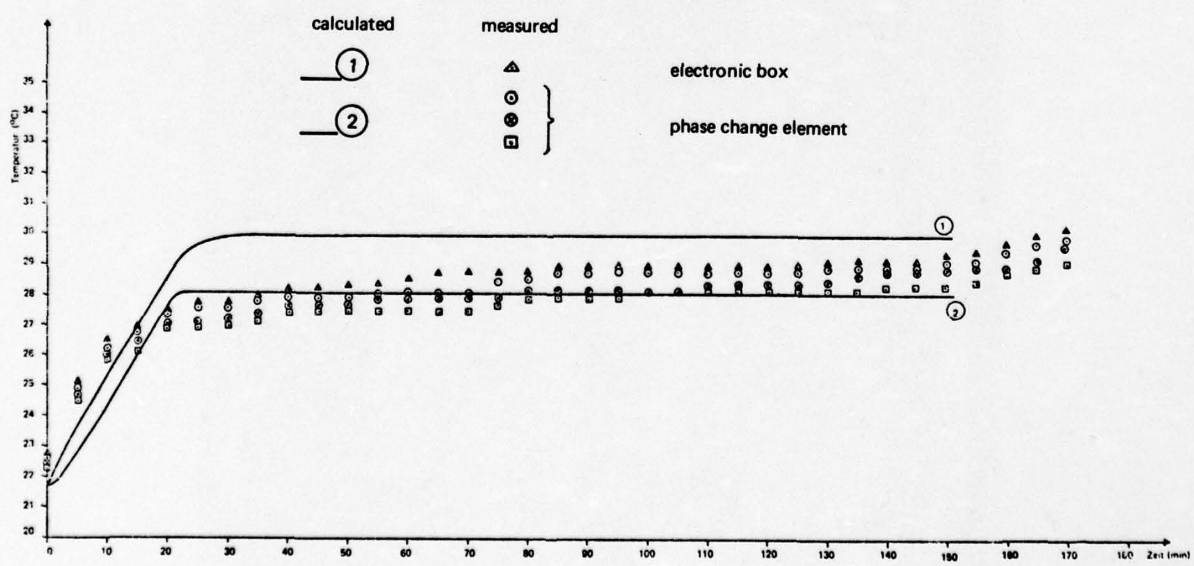


Fig.9 Temperature course of a phase change thermal capacitor

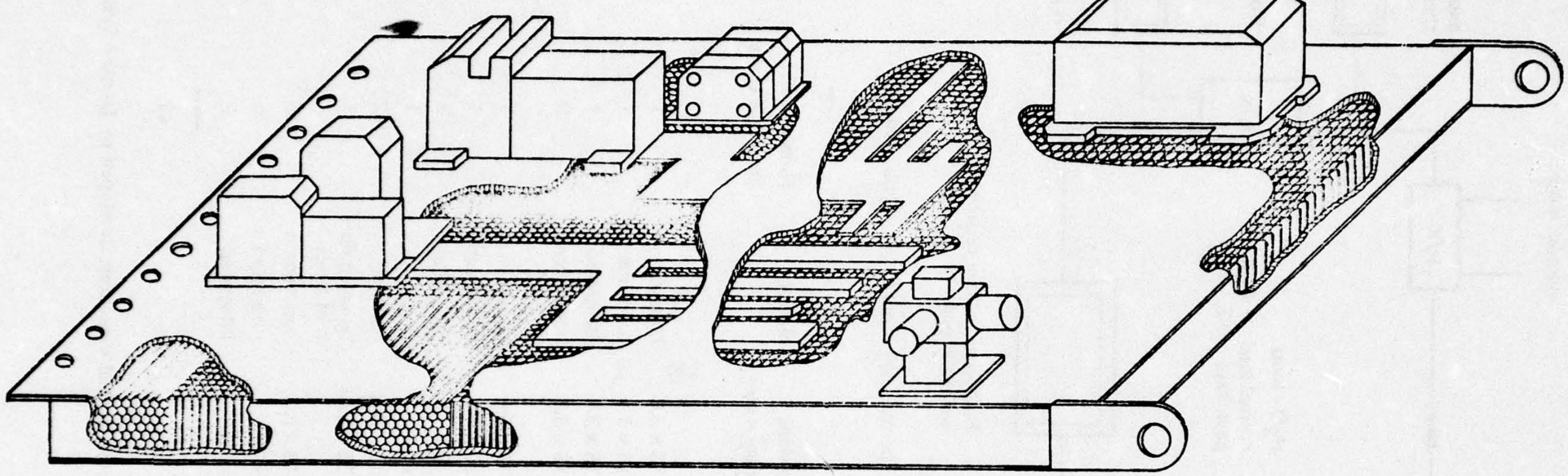


Fig.10 Phase change thermal capacitor mounting platform for heat dissipating equipment

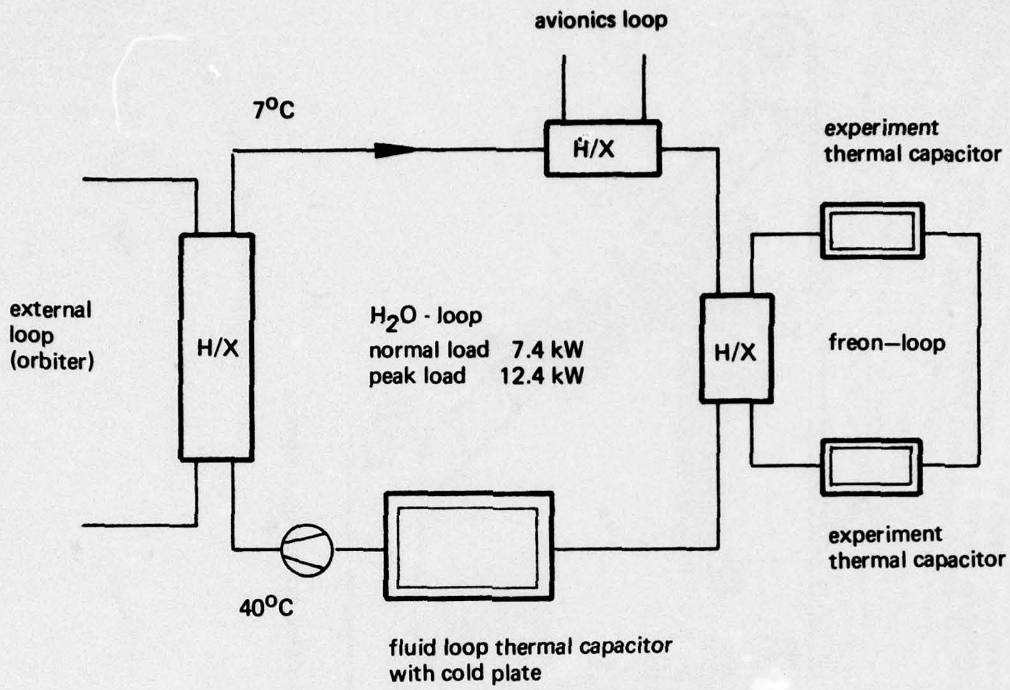


Fig.11 Thermal capacitors in S/L–cooling loops

Storage Capacity (Wh)	Operation Temperature (°C)	Dimensions (cm x cm x cm)	Matrix	Number of Produced Units	State of Development or Application
30	28	15 x 15 x 3.5	honeycomb	1	eng. model
30	28	15 x 15 x 3.5	h.c + fins	1	eng. model
25	28	15 x 15 x 3.5	integrated	1	eng. model
30	28	15 x 15 x 3.5	honeycomb	9	qualification life test
150	28	35 x 18 x 5.9	honeycomb	1	HLS-specification
380	28	50 x 50 x 3.3	honeycomb	1	ATC-specification
24	36	8 x 8 x 12	integrated	2	flight units
20	58	6 ∅ x 12	fins	1	eng. model
25	28	15 x 15 x 3.5	honeycomb	2	volume compensation
120	28	65 x 35 x 2.5	honeycomb + heat pipes	1	eng. model
30	28	15 x 15 x 3.5	honeycomb	15	qualification series
11	36	7 ∅ x 12	integrated	5	flight units
10	– 90	8 ∅ x 20	integrated	2	flight units
				42	

Fig.12 Phase change thermal capacitors, developed by Dornier System

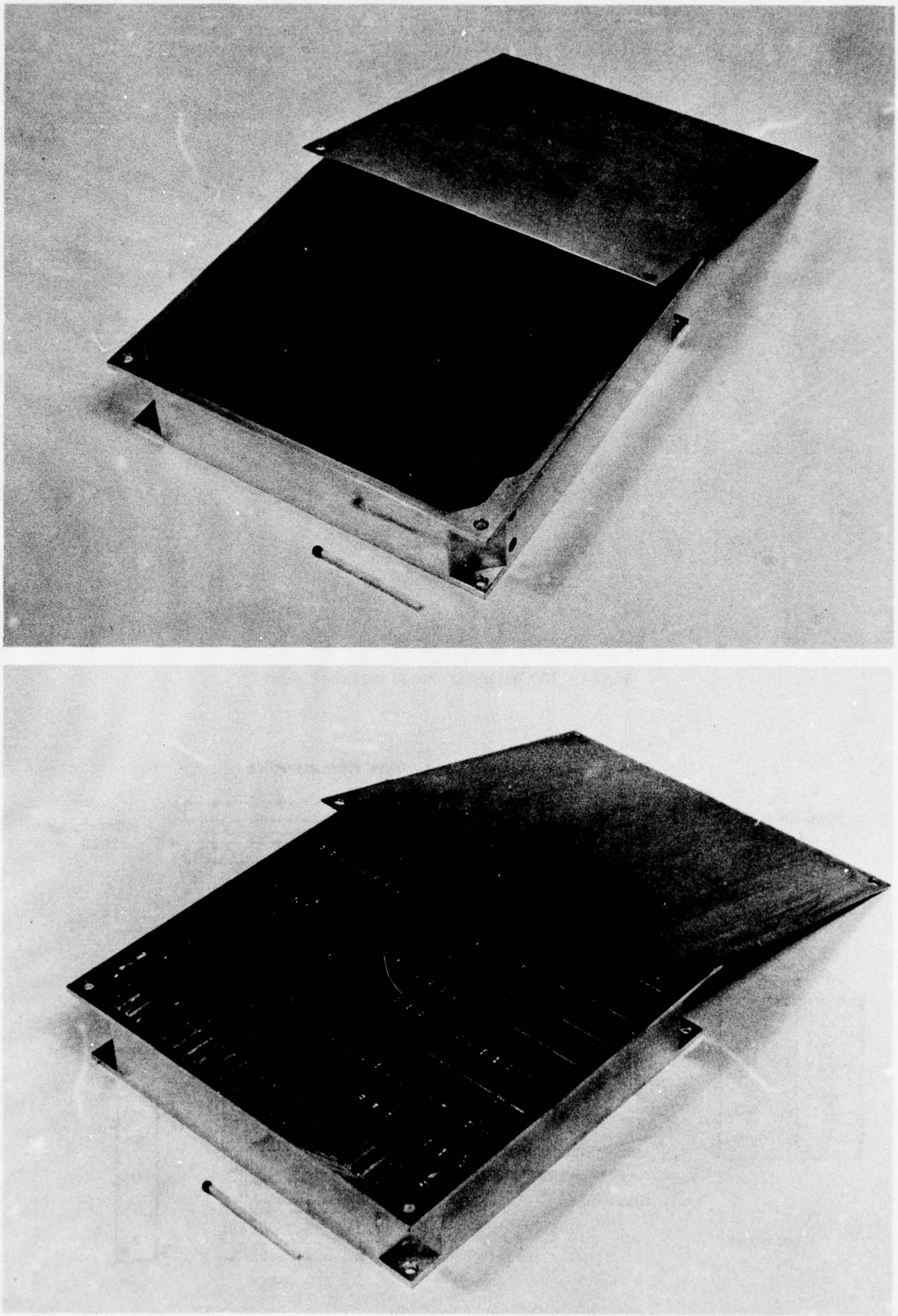


Fig.13 30 Wh Thermal storage element

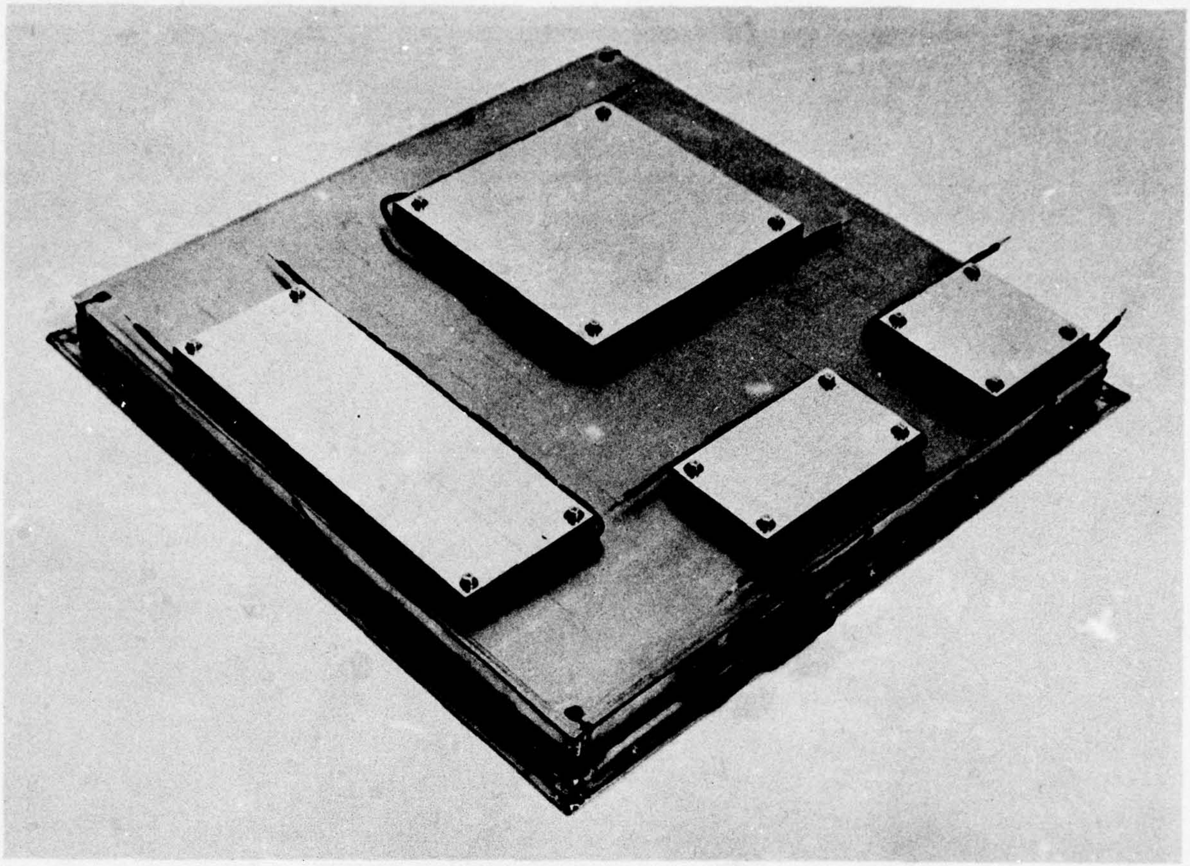


Fig.14 380 Wh phase change mounting panel

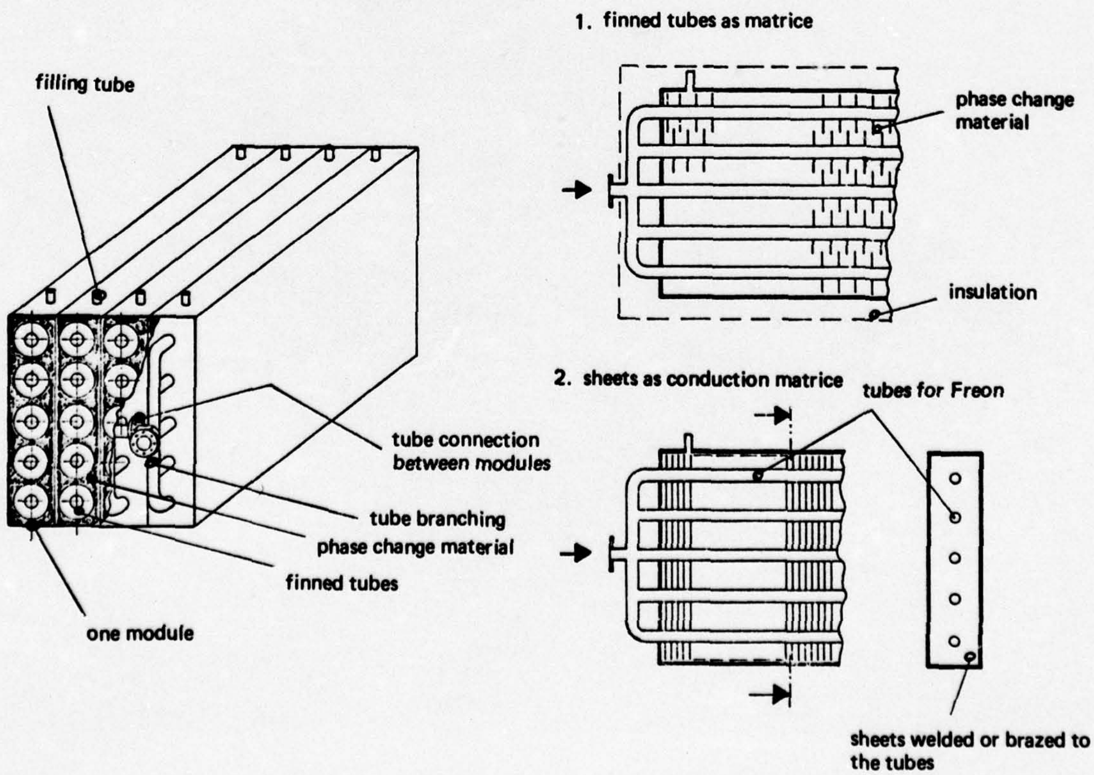
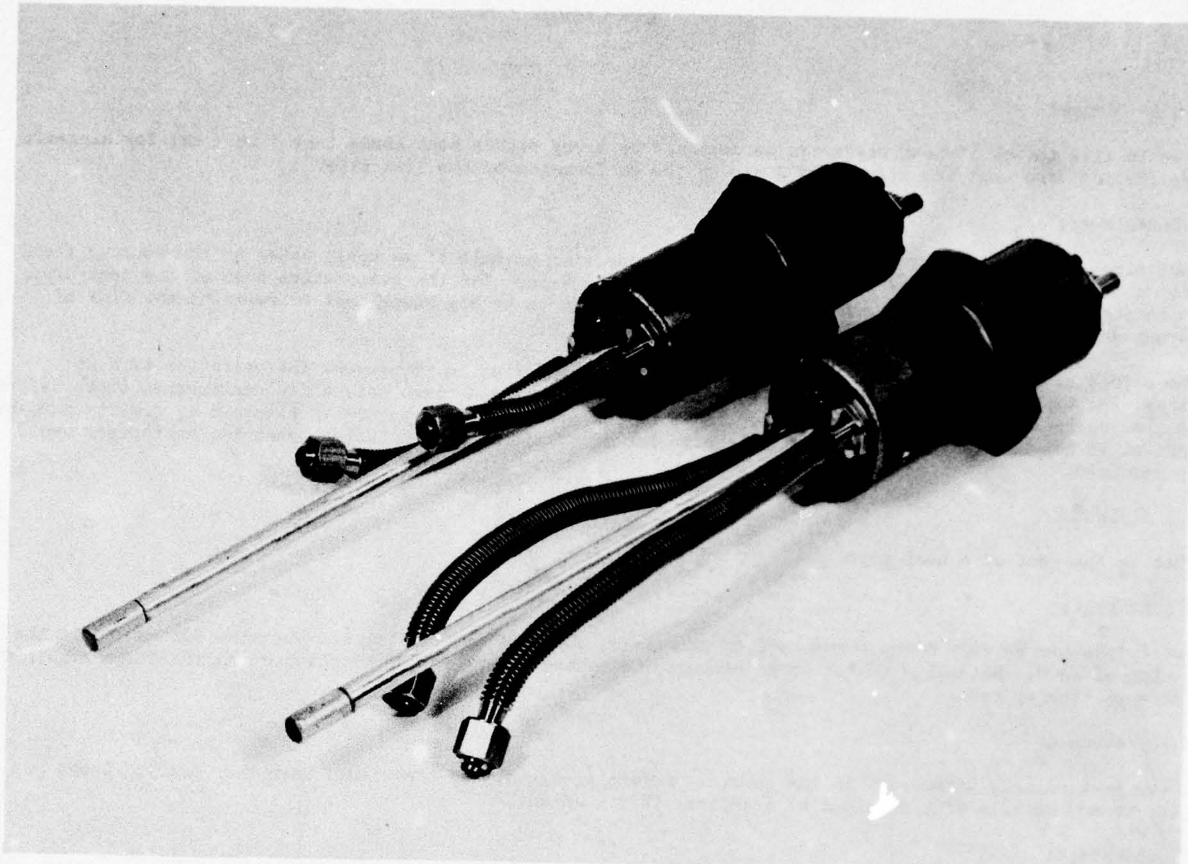


Fig.15 Cooling loop phase change thermal capacitor



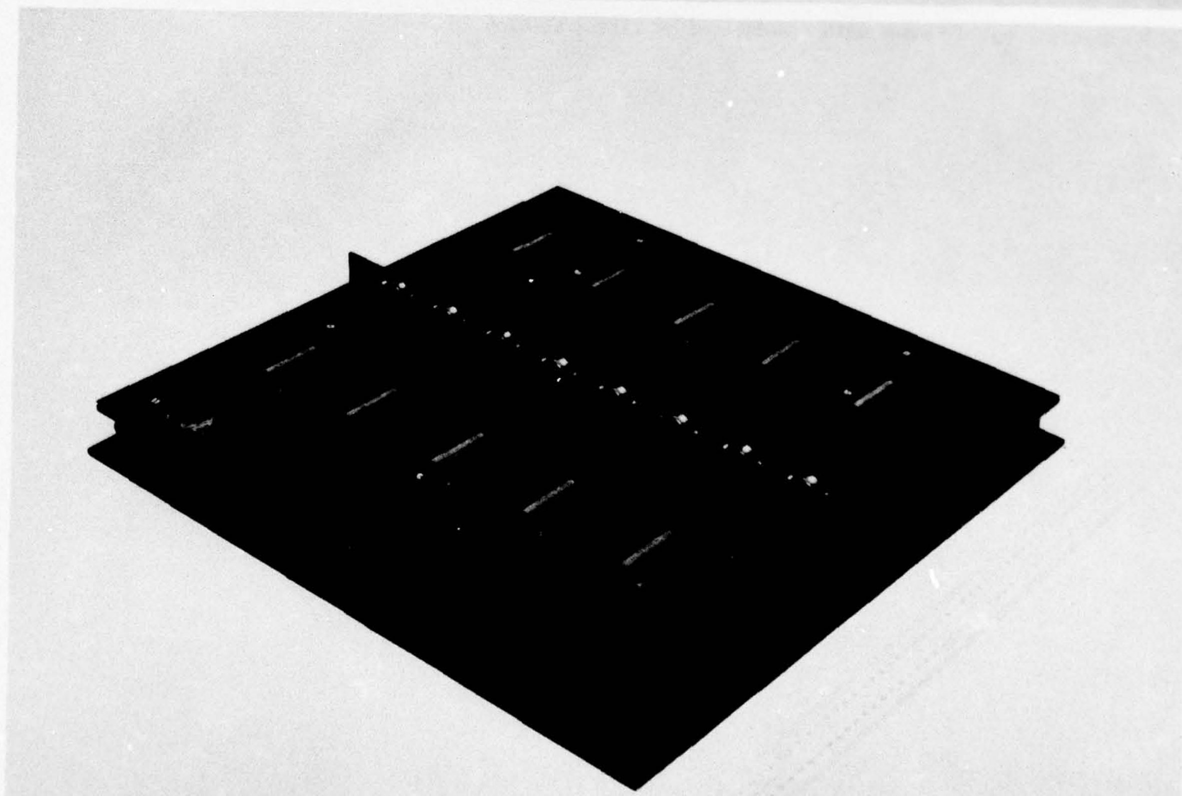


Fig.16/17 Phase change heat sink with fluid discharge and thermal capacitor for high dissipation electronics

DISCUSSION

R H Le Claire:

I would like to ask if heat pipes can be designed to carry higher heat loads (say 3 to 4 kW) for aircraft applications and what the effects of G are on the performance of the heat pipe?

W Schwarzott:

Heat pipes can be designed to carry higher heat loads, for example if we apply water as the working fluid with a heat transfer rate of 10,000 W-cm^{2}, we need only 4 cms^{2} for the evaporative zone of the heat pipe. Depending on the length of the heat pipe the diameter has to be big enough not to restrict the flow of vapour and liquid.

The effect of gravity when applied against the direction of flow is to prevent the operation of heat pipes, for example against the gravity forces of one g the liquid rises only a few centimetres (this refers to grooved heat pipes and wicked pipes). Under reflux operation (the liquid is returned by gravity forces), Heat pipes work without any problem. If the vector of acceleration fluctuates, then two heat pipes would be required.

F S STRINGER:

What is the cost of a heat pipe?

W Schwarzott:

Heat pipes can be very cheap. The cost of materials, fabrication, fitting and checking out can be in the region of $200. Naturally the price is strongly dependant on special application characteristics required for a particular task.

G F Stevenson:

I was particularly interested in the thermal storage elements, but they could be rather heavy. Could you give me any details of the weight of a typical thirty Wh unit?

W Schwarzott:

A 30 Wh capacity could be met with a weight of as little as 0.86 kgs.

CONCEPTION OPTIMALE DES EQUIPEMENTS ELECTRONIQUES AEROPORTES

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SOMMAIRE

De plus en plus d'équipements électroniques sont installés à bord des avions modernes, chacun de ces équipements nécessite une source d'alimentation électrique, et demande de plus en plus souvent un système de refroidissement.

Le refroidissement est nécessaire pour limiter l'élévation de température résultant de la dissipation d'énergie, et l'accroissement de la température ambiante provenant des vitesses supersoniques. La masse des systèmes d'alimentation électrique et de refroidissement est une fonction croissante de l'énergie dissipée.

Il est possible de réduire la masse des équipements aéroportés, mais si la puissance nécessaire n'est pas réduite, il en résultera un moins bon rendement, et une augmentation de la température ; par suite la réduction obtenue sur l'équipement peut être illusoire en raison de l'accroissement de la masse des systèmes d'alimentation et de refroidissement.

Par conséquent, il existe une solution optimale pour réduire la masse totale à sa valeur minimale.

Une méthode est proposée pour trouver cette solution optimale. Elle consiste à définir des relations entre tous les paramètres : dimension, puissance à l'entrée, puissance à la sortie, température, paramètres électriques tels que : Induction, pertes, conception des équipements ... La méthode montre en particulier comment optimiser une cascade d'étapes fonctionnelles.

Elle permet de déterminer une solution donnant les valeurs optimales de tous les paramètres, aussi bien que les besoins en énergie électrique qu'en refroidissement et des exemples d'application sont donnés.

La solution optimale fournit aussi des facteurs de mérite relatifs qui permettent d'établir une comparaison entre différents matériaux et de définir les améliorations souhaitables. Les calculs sont simplifiés en utilisant la technique de programmation dynamique. Les exemples donnés à titre indicatif concernent les conducteurs électriques, les transformateurs et les dispositifs de refroidissement à effet PELTIER.

La méthode proposée est générale et peut aussi être utilisée pour rechercher des économies de matières ou d'énergie, et est, par conséquent, d'un intérêt particulier à l'époque actuelle.

1. INTRODUCTION

Dans les avions modernes, les équipements électriques ou électroniques sont de plus en plus complexes et nombreux.

Les efforts de miniaturisation ont conduit à réduire les masses, les volumes, la puissance dissipée par élément et ceci a été possible grace à la microélectronique en particulier. Néanmoins l'accroissement du nombre des équipements et de leur compacité entraîne d'une part une élévation de la puissance électrique à fournir, et d'autre part, en raison de la puissance dissipée, une élévation de température qui peut nuire aux performances des systèmes et réduire en particulier la fiabilité.

Par conséquent, l'introduction d'un nouvel équipement consommant de l'énergie entraîne un accroissement corrélatif de la puissance dissipée.

Ce dernier accroissement, joint à l'augmentation de température qui résulte des vols à vitesse supersonique peut conduire à prévoir un dispositif de refroidissement ou de conditionnement.

Parmi plusieurs critères qui peuvent être retenus, la masse et le volume sont ceux qui intéressent le plus l'avionneur en général.

C'est-à-dire que pour réduire la masse des matériels électriques ou électroniques aéroportés, on peut être amené à faire des sacrifices sur le rendement. On augmente alors la demande d'énergie à la source, et par suite la masse de celle-ci. Pour savoir si, en définitive, un gain a été obtenu, il faut calculer la masse globale de l'ensemble : source plus équipement. Cette masse est appelée masse généralisée. C'est le critère qui permet de fixer les limites ultimes de la miniaturisation.

Pour des éléments dissipatifs, la réduction des dimensions, même sans diminution corrélative du rendement entraîne une élévation de la température de fonctionnement. Il peut en résulter des servitudes de refroidissement qui se traduiront par une masse supplémentaire de l'équipement proprement dit ou des organes annexes dont il faut tenir compte dans l'expression de la masse généralisée.

Par définition :

La masse généralisée est la somme de la masse propre de l'équipement considéré de celles de la source d'énergie et des dispositifs de refroidissement nécessaires pour en assurer le fonctionnement. C'est l'équipement de masse généralisée minimale qui donne à l'avion ou à l'engin la charge la plus légère.

2. PRINCIPE DE LA METHODE

2.1 Expression de la masse généralisée

Considérons un équipement de masse P_i qui assure une certaine fonction et absorbe une énergie électrique W_i à l'entrée et fournit à la sortie une énergie électrique W_{i+1} (fig. 1).

La masse P_i de cet équipement dépend de sa conception, mais aussi de W_i et de W_{i+1}. Pour marquer cette relation on la notera : $P_i(W_i, W_{i+1})$

La puissance dissipée dans l'étage $W_i - W_{i+1}$ constitue les pertes dans cet étage.

En général c'est la valeur W_{i+1} qui est fixée. Donc suivant la conception de l'étage i les pertes vont être plus ou moins importantes et la grandeur Wi va donc varier en conséquence.

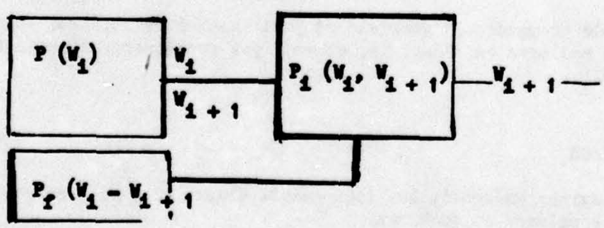
Cette variation va réagir sur la source qui fournit cette énergie, donc sur la masse $P(W_i)$.

De même l'énergie dissipée dans l'étage i, va entraîner une élévation de température, fonction également de la conception de l'étage et de l'importance des pertes et réagir sur la masse du dispositif de refroidissement qui variera suivant les pertes, et que l'on notera $P_f(W_i - W_{i+1})$

Il s'agit donc de rendre minimale la masse généralisée représentée figure 1 et définie par

(1) $$P_g = P_i(W_i, W_{i+1}) + P(W_i) + P_f(W_i - W_{i+1})$$

fig. 1



2.2 Relations masse-énergie-frigories

En général P_i est définie par la nature de la fonction à remplir et dépend de m variables x_m (2) $P_i = f(x_1, x_2, \dots x_m)$

De même Wi est fonction de p variables x_p

(3) $W_i = \varepsilon_i(x_1, x_2 \dots x_p)$

(4) $W_{i+1} = \varepsilon_i(x_1, x_2, \dots x_p)$

Ces variables sont en général des grandeurs physiques (électriques, thermiques, mécaniques) et des dimensions géométriques ; certaines sont communes aux W_i et P_i sinon il y aurait indépendance entre ces grandeurs.

En ce qui concerne la forme des relations $P(W_i)$ et $P_f(W_i - W_{i+1})$ peut admettre en première approximation que dans le domaine considéré, elles sont linéaires et de la forme (figure 2)

(5) $P(W_i) = A + GW_i$

(6) $P_f(W_i - W_{i+1}) = B + F(W_i - W_{i+1})$

G est le coefficient d'équivalence exprimé en grammes/watts pour la fourniture d'énergie électrique.

F est le coefficient d'équivalence exprimé en grammes/watts pour la fourniture de frigories.

Avec ces hypothèses, la masse généralisée s'écrit :

$$P_{Gi} = A + G_{Wi+1} + P_i(W_i, W_{i+1}) + B + F(W_i - W_{i+1})$$

ou encore en tenant compte que $GW_i = G(W_i - W_{i+1}) + G_{Wi+1}$

$$P_{Gi} = A + F + GW_{i+1} + P_i(W_i, W_{i+1}) + (G+F)(W_i - W_{i+1}) \qquad (7)$$

C'est l'expression de P_{Gi} qu'il faut rendre minimale.

2.3 Expression de la masse généralisée

La puissance de sortie de l'étage W_{i+1} est la puissance utile. Donc les termes $A + F + GW_{i+1}$ sont constants et ils n'interviendront donc pas dans la recherche du minimum de P_{Gi}. On appellera aussi masse généralisée que l'on notera P_{gi} la somme des termes :

(8) $P_{gi} = P_i(W_i, W_{i+1}) + (G+F)(W_i - W_{i+1})$

Autrement dit : la masse généralisée de l'étage i est égale à la masse propre de l'équipement et à la masse équivalente aux pertes dans l'étage

On voit ainsi que l'emploi du refroidissement forcé, avec les hypothèses revient à majorer le coefficient d'équivalence G de fourniture d'énergie.

3. CALCUL D'UN ENSEMBLE DE MASSE MINIMALE

3.1 Définition

Compte tenu de ce qui précède on voit que dans la plupart des cas les équipements peuvent être dissociés en une cascade d'étages (fig. 3).

Ces étapes représenteraient par exemple, le générateur électrique, les conducteurs, les transformateurs, les redresseurs, les filtres etc ...

fig. 3 $\boxed{P_o} - W_1 - \boxed{P_1} - W_2 - \boxed{P_2} - W_3 -$

Chaque étage i (i = 1, 2, 3, n) a une masse P_i et reçoit à l'entrée une grandeur W_i et fournit à la sortie une grandeur W_{i+1}.

On se propose de déterminer la suite des W_i et des P_i de telle sorte que la somme

$$P_o + \sum_{i=1}^{n} P_i \quad \text{soit minimale}$$

On suppose que la grandeur de sortie $W_{n+1} = W$ est connue ainsi que la fonction $P_o(W_1)$ qui représente par exemple la masse de la source (électrique et frigorifique) en fonction de la grandeur W et n'est fonction que de W_1.

La masse généralisée P_{Gn} est la somme des masses réelles (fig. 1)

$$P_{Gn} = P_o + P_1 + \ldots + P_n \qquad (1)$$

Considérons le cas des 2 premiers étages :

on a :

(9) $P_{G1} = P_o + P_1 = P(W_1) + P_1(W_1, W_2)$

Quelle que soit la grandeur W_2 considérée comme un paramètre, on peut déterminer toutes les variables propres à l'étage 1 de telle sorte que P_{G1} soit minimum. Cette condition détermine P_1 et W_1. Donc à toute valeur W_2 correspondent des valeurs P_1 et W_1 qui satisfont à la condition :

$$(10) \quad P(W_2) = \text{Min} \left[P(W_1) + P_1(W_1, W_2) \right]$$

Par récurrence, on a la relation :

$$(11) \quad P(W_n) = \text{Min} \left[P(W_n) + P_{n-1}(W_{n-1}, W_n) \right]$$

qui permet de déterminer la suite des W_i et P_i en séparant les variables propres à chaque étage.

4. EXEMPLE D'APPLICATIONS - CAS DES CONDUCTEURS

Le principe du calcul de la masse généralisée peut s'appliquer aux conducteurs, et en particulier aux cablages qui relient entre eux plusieurs appareils.

Le problème à résoudre est alors le suivant : on veut réaliser un cablage dont la masse généralisée soit minimale, sachant que les conducteurs doivent laisser passer des courants donnés.

L'application des méthodes générales permet :

- de classer les matériaux conducteurs dans l'ordre d'intérêt
- lorsqu'un matériau a été choisi, de l'utiliser au mieux.

4.1 Calcul de la masse généralisée du conducteur

Soit un conducteur homogène, de rayon r, réalisé en un matériau dont la résistivité ρ varie en fonction de la température suivant la relation :

$$(17) \quad \rho = \rho_0 (1 + \beta T)$$

ρ_0 étant la résistivité à 0 °C; et β le coefficient de température du matériau considéré.

Le conducteur a une densité : d_c

Considérons la longueur unitaire de ce conducteur. La masse réelle est :

$$d_c \; s$$

s étant la section utile $\quad s = \pi r^2$

La résistance de l'unité de longueur du conducteur est :

$$\rho / s$$

et les pertes dans le conducteur :

$$\rho \frac{I^2}{s}$$

Si on suppose que le conducteur est refroidi naturellement, et que le coefficient d'équivalence de la puissance fournie en grammes par watt est G, la masse équivalente aux pertes dans le conducteur est :

$$\rho \frac{G I^2}{s}$$

et la masse généralisée de l'unité de longueur est :

$$Pg = d_c s + \rho \frac{GI^2}{s} \qquad (13)$$

d_c et ρ sont fonction du matériau choisi. Ce choix étant fait la variable est s.

Le minimum de la masse généralisée est obtenu pour :

$$d_c s = \rho \frac{G I^2}{s}$$

ou

$$s2 = \rho \frac{G}{d_c} I^2 \qquad (14)$$

En faisant apparaître la densité de courant :

$$D = I/s$$

le minimum de la masse généralisée est obtenu pour la densité de courant optimale D_o

$$D_o = \sqrt{\frac{d_c}{\rho G}} \qquad (15)$$

On voit que la densité de courant optimale est fonction :

- de la densité du matériau
- de la résistivité du matériau et de la température réelle à laquelle fonctionne le matériau.
- du coefficient d'équivalence G.

Lorsque la densité de courant a la valeur optimale, la masse généralisée est donnée par :

$$Pg = 2\, d_c s = 2\, d_c \frac{I}{D_o} = 2\, I \sqrt{G \rho d_c} \qquad (16)$$

__Conclusion.__

Le cablage qui a la masse généralisée la plus faible pour des valeurs de I et G données, est celui pour lequel le produit $\rho\, d_c$ est le plus faible.

__Expression de la masse généralisée.__

Lorsqu'on prend pour densité de courant la valeur optimale, la masse généralisée est donnée par :

$$P_g = 2\, d_c \frac{I}{D_o} \qquad (17)$$

Cas où la densité de courant est différente de D_o :

Si on prend une densité de courant différente de la densité de courant optimale, la masse généralisée est donnée par :

$$P_g = d_c \frac{I}{D_o} \left(\frac{D_o}{D} + \frac{D}{D_o} \right) \qquad (18)$$

La courbe de la figure 4 représente la variation de la masse généralisée d'un conducteur en fonction de la densité de courant (rapport D / D_o).

Le minimum est peu accusé, et pratiquement il est donc possible de prendre des valeurs de D légèrement différentes de la valeur optimale, et compatibles avec les intensités utilisées et les diamètres de fils existants, sans perdre de manière sensible sur la masse généralisée.

4.2 - Classement des matériaux par ordre d'intérêt.

La relation (16) permet de dresser un tableau des valeurs de ρ et d_c de divers matériaux, à diverses températures de fonctionnement, ce qui permet de classer ces matériaux par ordre d'intérêt décroissant, par exemple.

Pour des matériaux qui ont sensiblement le même coefficient de température, il suffit de les comparer à une même température. Pour des matériaux qui ont des coefficients de température différents, il sera nécessaire de faire la comparaison aux températures réelles d'emploi, qui peuvent ne pas être les mêmes pour deux matériaux employés dans les mêmes conditions, les conditions de refroidissement étant fonction de la surface latérale du conducteur, donc de son diamètre.

Le tableau 1 donne les valeurs de ρ, d_c, et $\sqrt{\rho\, d_c}$ pour divers matériaux.

Ce tableau montre que les matériaux se classent par ordre d'intérêt décroissant comme suit :

aluminium - magnésium - cuivre - argent.

Des considérations autres que celles de masse généralisée peuvent intervenir dans le choix du conducteur. Cependant, cette notion de masse étant considérée comme importante en aéronautique, les résultats précédents permettent de connaître le facteur de qualité : $\rho\, d_c$. Le matériau le plus intéressant au point de vue masse est celui qui a le produit $\rho\, d_c$ le plus faible.

4.3 - Calcul des chutes de tension.

La chute de tension par unité de longueur est donnée par :

$$V = R\, I = \rho \frac{I}{s} = \rho\, D \qquad (26)$$

La chute de tension par unité de longueur est directement proportionnelle à la densité de courant.

4.4 - Relation densité de courant-température

On applique la méthode générale indiquée précédemment en tenant compte des conditions réelles de refroidissement.

5. - CONCLUSIONS

La méthode a été appliquée aux transformateurs, inductances, filtres et dispositifs à effet Peltier [1] et [4].

Mais ce qui est peut être le plus important est que la méthode permet d'établir un langage commun entre l'avionneur qui a la responsabilité globale de la conception et du bon fonctionnement de l'aéronef et le constructeur d'équipement à qui l'avionneur impose des contraintes et en reçoit aussi.

La solution optimale ne peut être qu'un compromis, mais le meilleur compromis est celui qui s'appuie sur des données claires, acceptables par tous.

En d'autres termes on peut dire que l'un des problèmes à résoudre consiste à bien définir l'interface entre avionneurs et fournisseurs d'équipements.

Les paramètres qui interviennent dans la définition de ces interfaces sont :

- les caractéristiques des sources d'énergie : tension, fréquences, impédance, et variation de ces caractéristiques, stabilité du réseau etc.
- environnement, température, pression suivant les diverses phases de vol.
- conditionnement, nature du refroidissement naturel, forcé(à air liquide,) température de source froide, capacité calorifique, etc.

Certains de ces paramètres sont fixés par l'avionneur, mais celui-ci a besoin de connaître les besoins des équipements pour établir son projet final.

La méthode proposée partant du concept de masse généralisée minimale, permet à l'avionneur au stade du projet de faire connaître les paramètres importants tels que caractéristique de l'alimentation électrique, température de la source froide, débit du fluide de refroidissement, coefficient de masse en grammes par watt, (évacué ou fourni). Plusieurs séries de valeurs peuvent être soumises.

Ces paramètres permettent à l'équipementier de définir ses équipements pour la masse généralisée minimale, éventuellement pour les diverses séries de valeurs.

Cette méthode permet donc d'obtenir une excellente définition des interfaces avec un critère commun et d'arriver ainsi rapidement au meilleur compromis.

Un des problèmes qui se pose, par exemple, est celui de savoir si l'avionneur doit fournir une alimentation à fréquence et tension très stabilisée ou non.

La réponse à cette question n'est pas évidente et dépend peut être des applications - Arriver à un accord peut être très difficile.

Le concept et la méthode proposés permettent de faire des simulations avec des hypothèses différentes et de comparer les résultats entre eux. Ainsi le choix de la solution de compromis devient scientifique et se fait sur des bases solides.

Améliorer globalement les performances, faciliter les relations avionneurs-constructeurs d'équipements pour y arriver sont les résultats attendus de l'application de la méthode.

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TABLEAU 1 : Coefficient de mérite de divers conducteurs électriques. Les valeurs les plus faibles sont les meilleures.

CONDUCTEUR	d_c / g	ρ (20°ρ) Ω-cm	$\sqrt{\rho d_c}$
ALUMINIUM	2,7	2,83	2,76 10-3
MAGNESIUM	1,74	4,6	2,83 "
DURALUMIN (AU4G)	2,8	4,5	3,55 "
CUIVRE	8,9	1,724	3,94
ARGENT	10,5	1,64	4,15
OR	19,3	2,44	6,87

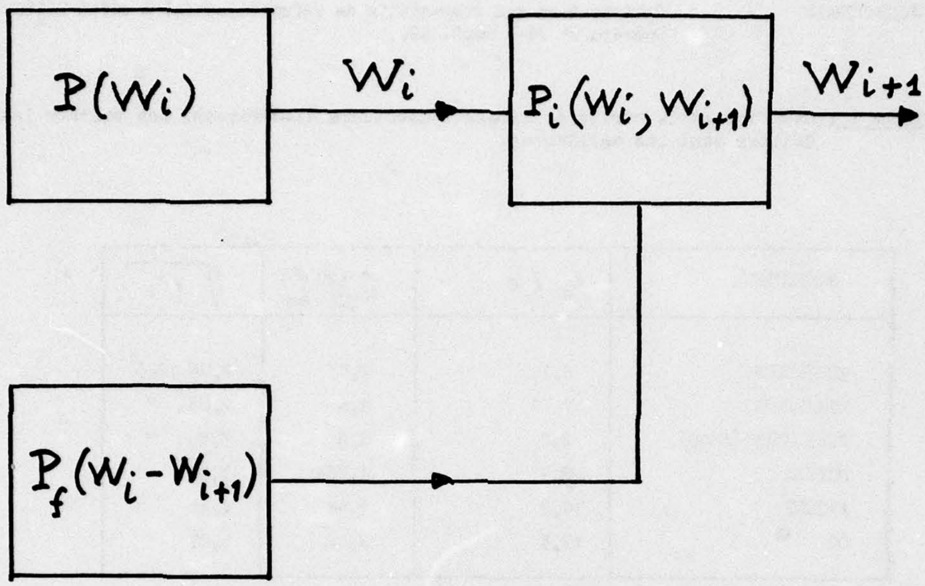

Figure 1 : **Masse généralisée d'un équipement de masse propre P_i, consommant une puissance W_i et délivrant une puissance W_{i+1}, alimenté par une source d'énergie de masse $P(W_i)$ et refroidi par un système de masse $P_f(W_i - W_{i+1})$**

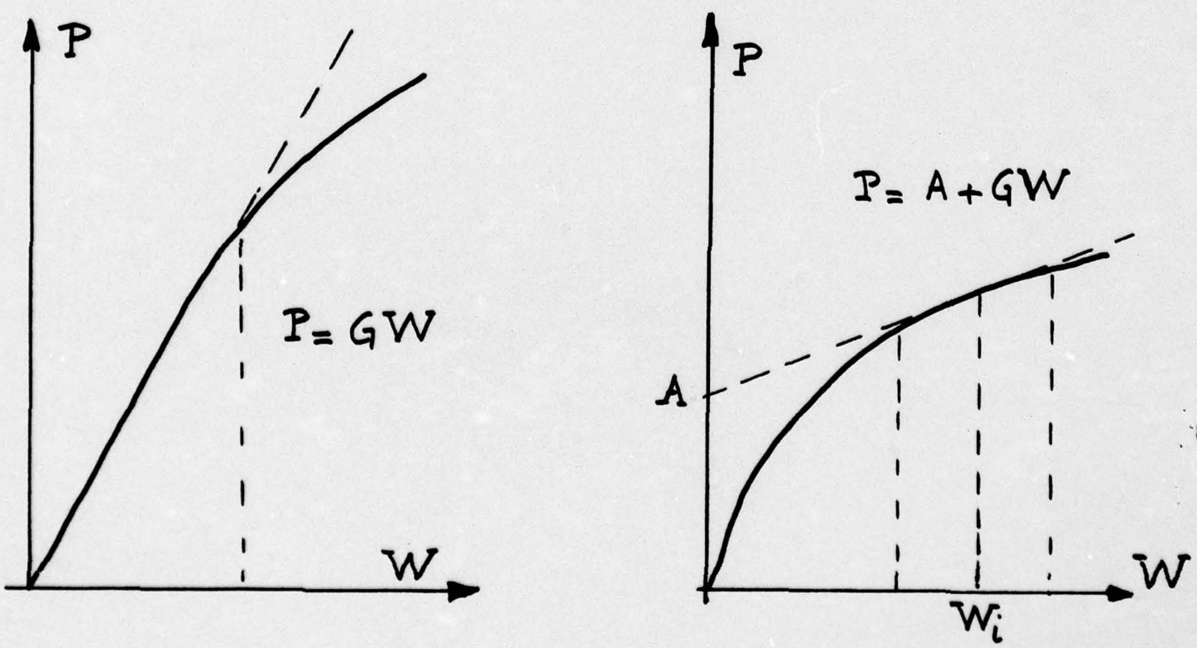

Figure 2 : **Modèles mathématiques représentant la masse des systèmes d'alimentation électriques en fonction de la puissance fournie.**

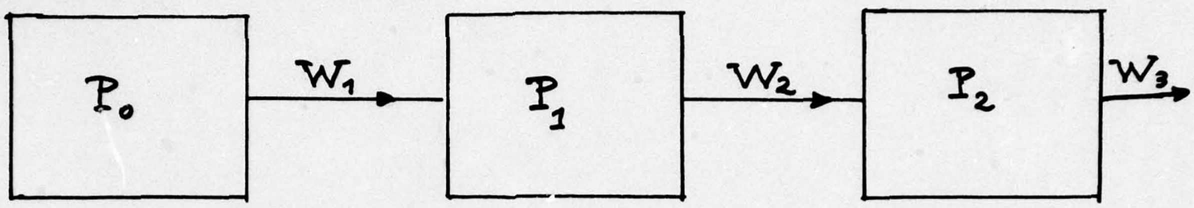

Figure 3 : **Cascade d'étages fonctionnels.**

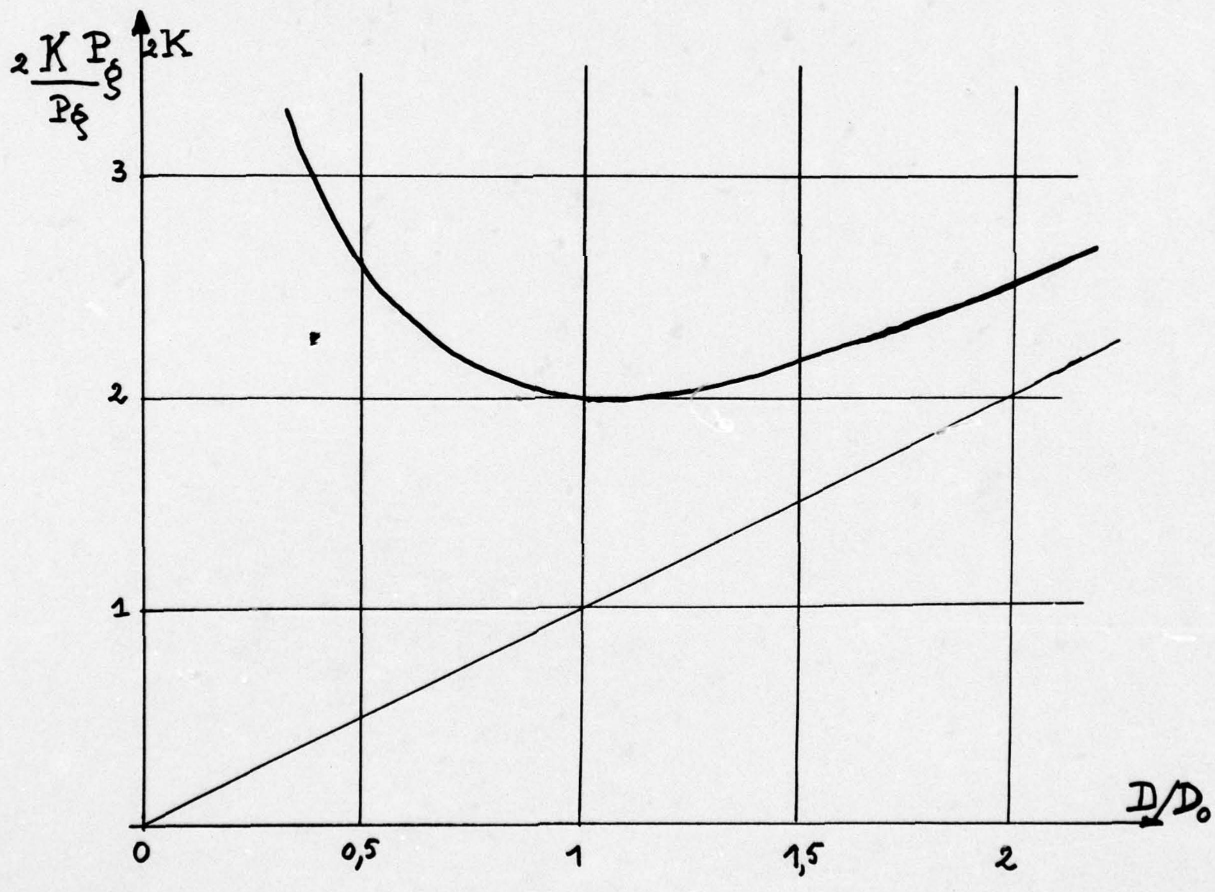

Figure 4 : **Variation de la masse généralisée lorsque la densité de courant des conducteurs n'est pas optimale.**

DISCUSSION

G Borgonovo:

What is the relationship between the optimisation of cooling watts and the optimisation of price?

J Bertrais:

This relationship has not been investigated. However it seems likely that the minimum of equipment cooling capacity will coincide with the minimum cost, as long as simple equipment is dealt with.

THE POSSIBLE IMPACT OF DC AIRCRAFT POWER SUPPLIES ON THE DESIGN OF AVIONIC AND OTHER EQUIPMENT

by

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1. DISTRIBUTION CONTROL

Power consuming equipment in aircraft generally falls into two types – that which uses power directly (e.g. de-icing, galley equipment, fuel pumps, etc.), and that which processes the power before use (e.g. instrumentation, flight control).

Whatever the load requirements are however, isolating devices must be installed in the feeds to all loads. These devices must be able to isolate loads whatever the load fault may be. Such devices are well defined and understood so far as Alternating Current supplies are concerned. In the case of DC however, one problem in isolating a load is over-riding. This is the so-called uncontrollable arc. Indeed, if it were not for this phenomenon, DC could well have been used universally for power distribution. Consideration therefore of means to isolate DC circuits is paramount.

Methods available are mechanical contactors, thyristor switches and transistor switches. The range of loads to be switched in a typical fighter aeroplane are from about 20 watts to 3.5 Kw with more than 70% below 250 watts. In transport aircraft, the proportion is not very different.

A NASA specification for a 120 volt DC Power Controller up to a 30 A load has been shown to be feasible. This switch will handle up to a 200 volt overload with a 90 Ampere fault protection. Furthermore, the switch is easily controlled remotely by a 15 volt DC control signal. In normal use, the voltage drop across the switch will not exceed 1 volt. This represents 30 watts maximum dissipation, although it can be expected that switches can be fabricated to give 0.5 volts saturation of the transistor which reduces the power loss to 15 watts. For loads therefore up to say 30 A volts, remotely controlled power switches are feasible. It can therefore be concluded that it is worthwhile to examine the effect that DC power supplies can have on equipment design.

Many equipments use DC of varying voltages. At the present time, these DC voltages are derived from 400 Hz AC via transformer – rectifier circuits. In modern equipment, DC supplies often need to be stabilised, and, in the case of much digital equipment, the output impedance of the supply needs to be low over a wide frequency band. The most efficient means of producing such supplies is by the so-called 'chopper-stabilised' system. These systems are essentially variable mark-space switches basically handling power. Because of this they are intrinsically of high efficiency, providing that:–

(a) The transient response of the switching device is much faster than the chopping frequency.

(b) The input voltage to the switch is high.

With modern semiconductor devices, both of these criteria can be met. The resulting arrangement can be described as a DC-DC transformer.

This system does have another attribute which is that the input voltage can vary over a very wide range, because it is a power switch. Thus a system normally using 120 volt input can be switched to 28 volts without change in components or chopper frequency and incurring the penalty of only a small loss of efficiency. Such an attribute makes it easy to switch vital equipment to any available DC supply.

It is also very easy to parallel DC supplies before stabilisation.

The frequency at which these supplies operate can produce Radio Frequency Interference. This, by careful design, can be reduced to negligible proportions outside the equipment box because it is high, typically above 20 Kilo Hertz. Generally this has to be properly suppressed to avoid spurious switching voltages inside the equipment.

The use of high DC input voltages helps to reduce the problem because the line currents are smaller due, once again, to the fact that power switches are being used. In conditions where an input supply has to be changed from 120 to 28 volts, for example, any RFI may marginally increase, but such a situation will usually mean that much other equipment will not be in use, and hence the problem is unlikely to be significant. It may be mentioned that the minimum frequency of 20 Kilo Hertz is very unlikely to interfere with voice communication.

2. WEIGHT AND HEAT EFFECTS

2.1 Equipment

Many avionic equipment units carry their own stabilised supplies. A census of these produced in one company shows that between 27% and 38% of the weight of the units is directly due to the power supply. Of this, about 33% is due to transformers. Hence, about 10% of the weight of the units is solely due to the transformers.

Overall efficiencies of power supplies are around 70%. Thus, 30% of heat is contributed by the power supply, and of this 30%, some 25% is contributed by transformers. A 7½% saving in heat is therefore possible by eliminating the transformers. A further reduction in heat can be obtained by eliminating the rectifiers on the transformer secondaries. Heat savings from this cause are very dependent upon the initial DC voltage. The higher the voltage, the lower the dissipations – due to the constant voltage drop of about 0.5 volt across rectifiers. Savings of the order of 10% of power supply dissipation or 3% of the total heat in the units is about the maximum that can be expected from this source. The total heat saving therefore by eliminating transformers and rectifiers will be about 10%.

There is an argument for eliminating transformers on incoming AC supplies as is done in an ever increasing number of commercial equipments. However, this has two major shortcomings. The first is the danger of generating DC on the AC bus in the event of rectifier failure, the second, the generation of radio frequency interference by the diode switching action, which is difficult to suppress when directly on line.

2.2 Motors

Motors used in instrumentation and Flight Control are generally AC motors of the induction or hysteresis type. These motors are very suitable for analogue systems but the advent of digital techniques has increased the use and attractiveness of the stepping motor in control systems. In many cases, the use of stepping motors reduces the power consumption because of the torque available from them at zero current. As is well known, AC motors of conventional types require continuous power input to produce torque.

A further advantage of the stepping motor is the fact that its position and speed can be tightly controlled by the input pulses. There is no fundamental requirement for any feedback device. This saves both power and weight. Recent advances in the design of the variable-reluctance and hybrid motors with steps as small as 1.6° make them even more attractive. Strictly speaking, the use of stepper motors is in no way dependent on DC power distribution systems, since they are already in use on AC distributed systems, but in themselves they need no AC energisation.

Gyro motors are almost universally of the hysteresis type. While mechanical gyros remain in use, there seems no alternative to hysteresis motors, and hence DC-AC invertors will be necessary for producing AC power. There is an advantage in this procedure, particularly where digital systems are in use since very accurate frequencies can be generated giving corresponding improvements in gyro output accuracy. No major difficulties from the use of non-sinusoidal waveforms have been experienced.

Development of other types of gyros (e.g. Laser Gyro) is not affected by whether primary power is AC or DC.

It is not the purpose of this paper to consider high power continuous loads such as fuel pumps and air conditioning equipment, but it should be said that the AC induction motor has really no rival in these applications. AC supplies for these motors can be generated by modern technology without undue difficulty providing the motors do not suffer from loss of power or unacceptable heating when powered by square rather than sine wave AC.

As development proceeds the brushless DC motor will become more reliable and will show a better power weight ratio than AC motors because of the use of rare-earth and/or ceramic magnets. It can therefore be expected that eventually DC motors will be available for all aircraft tasks. In small aircraft and for some less important jobs, the permanent magnet DC commutator motor will still be used.

2.3 Wiring

Because only two wires are required for distribution, there is a self-evident saving which again will depend on the installation. Various estimates have been given for military aircraft. Figure 1 shows the comparison between various methods of power generation and distribution for a fighter aircraft. Similar comparisons for a transport aircraft should show substantial savings.

Weight Comparisons 2 Channel 60 KW System

	IDG	VSCF	CFG	HVDC
Power Source	213	299	257	147
Power Feeders	34	50	34	27
Power Conditioners	32	32	32	45
Distribution Wiring	700	700	700	350
TOTAL	979 – 569	1081	1023	569
	400 lb Weight Saving			

IDG Constant speed drive and AC generator (integrated)

VSCF Variable speed generator and power convertor for constant frequency AC

CFG Variable speed DC generator for constant frequency AC

Fig.1 High voltage DC power

3. CONSIDERATION OF THE IMPACT OF DC POWER DISTRIBUTION ON HIGH INTEGRITY SYSTEMS

High integrity systems require high integrity power supplies. The use of DC allows simple switching methods where power failures may occur. The generator, be it a rotating machine, transformer-rectifier, or battery, is only critical in respect of polarity. Frequency and phase requirements do not exist. Where a number of units comprise parts of a high integrity system, the DC power to drive one or more units is often generated and switched from another unit. This practice, while essential to the system operation, puts the supply for two or more units at risk, if any unit causes an overload on the supply. A DC power system will ensure that circuit breakers of high reliability will be available to prevent damage to the unit supplying power if the failure is elsewhere. At present fuses are sometimes fitted to prevent undue damage, but they are often difficult to get at, of wider tolerance than desired and *untestable*. For example, a fuse can have been stressed to a point where it will blow for a maximum normal operating load. Such a phenomenon alters the risk level of the system. A self-resetting circuit breaker is easy to test and after testing the risk of failure in the system is unaltered.

A further advantage of DC power in high integrity equipment, and particularly in vital equipment, is the ease with which almost any voltage can be accommodated. As mentioned earlier, equipment could be designed to operate off any input from 120 volts (or higher) down to 28 volts (or even lower) with little loss in efficiency.

4. GROUND POWER SUPPLIES

Ground Power Supplies for AC have generally been rather less well controlled than Aircraft supplies, and this sometimes causes problems on ground test – particularly where a multi-channel system normally using separate aircraft generators is being tested.

If the demand is for DC power only, then a high quality supply is comparatively easy to produce because of battery stabilisation.

5. SAFETY

There are two major aspects to safety in the use of DC. They are:–

(a) The possibility of long period arcs causing fire.

(b) Electrocution because of the 'freezing' effect on human muscles by DC shock.

It is clear that both of these must be taken into consideration when a high voltage DC system is being contemplated.

6. CONCLUSIONS

In most cases, the use of DC for primary power in equipment will:–

(a) Reduce weight by the order of 10% – overall benefits in weight saving including distribution wiring might well be much higher.

(b) Reduce dissipation by the order of 10%.

(c) Give more flexibility in switching power supplies.

(d) Possibly reduce electro-magnetic radiation by the elimination of power rectifiers and by removing restrictions on some filter components necessitated by 400 Hertz supplies.

To achieve the maximum benefits in the aircraft, the following are necessary:–

(a) A range of DC circuit breakers, the specifications of which are already written and flexibility demonstrated.

(b) Further development of invertors for AC gyro and similar motors to give high efficiency overall.

(c) Development of invertors for Induction motors or brushless DC motors to drive continuously running pumps and fans.

(d) Development of reliable low dissipation DC-DC converters, particularly using Large Scale Integrated circuits to reduce size and cost.

ACKNOWLEDGEMENT

The author would like to thank his colleagues for their supply of information and the directors of Smiths Industries Limited for permission to present the paper.

There is a vast range of literature on stepping motors, brushless DC motors, AC induction motors, etc, which can be consulted for further information.

In addition, considerable work has been published under NASA auspices on the design of DC circuit breakers to MIL SPEC requirements MIL-P-81653.

DISCUSSION

K P Gerrity:

I would like to make the following observations on your paper.

Firstly, you base much of your case for a high voltage dc supply on the use of solid state switches which as yet cannot compete with electromechanical switchgear. In particular these electronic switches have much higher dissipations, you quote a 30A device with a V_{CE} of 1 volt giving 30W dissipation, compared with a contactor of 100mΩ contact resistance giving 3W (drive power say 100mA at 28V dc or 2.8W). Also they have large component counts with consequent high cost and low reliability.

Secondly, with reference to your comment on driving dc back into the supply, and the question on tolerance of this by national power supply authorities. I would like to emphasise that such practice can cause problems and the UK Central Electricity Generating Board have introduced fairly tight controls over the generation of such current distortion by consumer equipments.

Thirdly, in answer to your query 'Why are high voltage dc generators lighter (per kVA) than equivalent ac generators?' I would point out that the dc generator is a frequency wild ac generator plus rectification and this does not require a CSD, so saving its weight (the ability to generate over say a 2:1 frequency range does impose a mass increase on the generator but this is much less than the mass of a CSD).

Fourthly,you say that 'fuses are a pest' and suggest the use of electronic switchgear instead. Might I point out that many development electronic switches have to include fuses for safe fault clearance!

N L Sigournay:

Some main circuit breakers may well remain electro-mechanical in initial stages of development of HV DC supply systems. I think the real gains will come from breakers on the many low power loads of 250 watts and less.

Thank you for your comments on the back-driving of DC into AC supplies. Rules for limiting this need to be drawn up.

The fact that some development electronic switches use fuses for safety doesn't make fuses 'less of a pest'. Ways for circumventing their use must be developed.

A Templeton:

High current, high voltage devices have a current failure mode that is basically a hot spot problem. This is made worse when switching on as there is a current spreading problem. It is not true to say that the failure mode is predominantly a voltage mode.

N L Sigournay:

The remark was intended to apply to chopper transistors, not high current, high voltage devices. It is agreed that the high current, high voltage devices used in solid-state circuit breakers are not suitable for 'chopper-stabilised' power supplies where high switching speeds are required.

Newton:

Will the author please define what size of system he is considering. There is little, if any, prospect of a high voltage DC primary generating system because of the problem of switching hundreds of amps and of obtaining real isolation. The level of switches discussed are really associated with specific loads not total power. The heat dissipation involved with this type of switching element is markedly higher than with existing types and will probably involve the airframe designer providing heat sinks to cool the switches.

The data on comparative system weights, Fig 1.is somewhat out of context. The information I believe refers to a very large system where most of the feeders were proposed at 400V DC, hence the low cable weight. For similar power and voltage levels there is not a lot of weight saving between the 3 phase and dc configurations.

N L Sigournay:

The system was of a light-weight fighter using two systems of 270V DC at 60kW per system.

It is not suggested that DC power is ready and available now, but that there are many advantages in its future use when technology has improved sufficiently to provide the necessary control devices.

N F J Allum:

The use of 147 V DC is shown in your graph. I suggest that the overall savings shown overlook the losses either due to use of 150 V DC battery or to conversion equipment from conventional supplies. Hence these benefits are very illusory.

N L Sigournay:

The 147 is the weight in lbs of the HVDC generator. The suggested DC voltage for practical purposes is around 150 V. 70% of aircraft loads are less than 250W which only require a 3 amp device for the circuit breaker, of which many are available from different manufacturers. The benefits of chopper-stabilised supplies are not illusory because many are in use already.

AIRCRAFT POWER SUPPLIES - THEIR PERFORMANCE AND LIMITATIONS

by

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SUMMARY

A considerable proportion of the heat dissipated by avionic equipments arises in their power supply conditioning stages. Such conditioning is necessary because of incompatibilities between the requirements of avionic equipments and the quality of the aircraft power supplies.

The ideal solution is to generate a supply of the required quality but practical limitations on equipment prevent this; instead we must live with the best performance currently achievable.

This paper examines the causes and rates of occurrence of such effects as abnormal or emergency limits, transients and interruptions which raise the need for power supply conditioning. In the course of this examination, the contributions to the overall supply quality of each of the major generation system components and their mode of interconnection are highlighted.

From this breakdown it is hoped that avionic equipment and system designers will gain a better working knowledge of the performance and limitations of aircraft electrical power supplies, thereby avoiding excessive dissipation by overdesign.

1. INTRODUCTION

This paper deals with the basic problems of incompatibility between avionic equipments and electrical generation systems on modern military aircraft. Before continuing it should be stressed that in general there are not many incompatibilities which cause a direct malfunction of equipment, in fact many do not even come to light because the avionic equipment manufacturer takes the necessary steps to overcome possible problem areas. However, these 'unseen' incompatibilities can be just as bad as, or worse than, the ones in open evidence in that they cause penalties in cost, mass and perhaps most important heat dissipation.

Let us now define what constitutes an incompatibility. Basically, avionic equipments prefer an interruption and transient free a.c. supply of constant frequency and pure waveshape or ripple free d.c. Similarly, the generation system prefers loads which draw constant undistorted currents. Any deviation from these ideals represents a potential incompatibility. The most obvious deviations are as follows:-

a) the generation system provides a voltage which has transients, interruptions, a fairly large spread in voltage and on a.c., a large spread in the base frequency plus a fairly significant harmonic content.

b) many avionic equipments draw their power in current pulses which, due to generator and feeder impedances, induce harmonics into the bus bar voltage waveform. Also normal load switching and fault clearance cause transients on the voltage waveform, unbalanced loading causes individual phase voltages to deviate from the controlled average.

This, then, is the problem and in essence it is not a new one rather it has come to the fore with the rise in proportion of avionic equipments, which are inherently more sensitive to voltage and frequency variations than basic electrical loads such as heaters.

Unfortunately, the problem is likely to get worse as the quantity of avionics rises and since the outcome of the incompatibilities is usually a high level of dissipation a fresh look at the problem is called for.

It is the aim of this paper to examine the operation of typical generation systems and explain why and when deviations from the ideal supply occur and, where possible, how we can overcome them. It is not the purpose of this paper to offer solutions from the avionic equipment side of the problem, therefore references to such equipment are only made where they add to the coherence of the paper.

2. THE PROBLEM AREAS

Before proceeding to describe the component parts of a typical generation system we must first outline the known main problem areas. Basically, supply problems can be split into two categories viz. voltage and frequency.

2.1 Voltage Problems

2.1.1 The a.c. system voltage Ideally, this is a pure 400 Hz sine wave of constant R.M.S. voltage; in fact it suffers from the following deviations:

a) normal, abnormal and possibly emergency steady state limits

b) modulation

c) transients and interruptions

d) phase mutual displacements other than 120° on 3-phase systems

e) the presence of harmonics

f) unbalance between phases.

2.1.2 The d.c. system voltage Ideally, this is a pure constant voltage but in practice suffers from the following:

a) normal, abnormal and usually emergency steady state limits

b) ripple

c) transients and interruptions.

2.2 Frequency Problems

The following effects are characteristic of the a.c. system base frequency:

a) normal, abnormal and occassionally emergency steady state limits

b) transients and possibly beating or absence of phase locking on multichannel systems

c) modulation.

2.3 The Effects of these Deviations

Electronic equipment requires a constant or fairly constant voltage supply, so equipment designers generally include a voltage regulator to cut off all voltage excursions above the bottom limit of the widest band over which they are required to operate. Thus, whenever the system provides a voltage higher than the minimum, which is most of the time it is operating, the equipment regulator will be functioning and dissipating heat. Chopper-type regulators dissipate less heat than series-type regulators but a very large proportion of equipments draw relatively low power and from their individual point of view the series type regulator is more appealing with the result that system dissipations are high. This is not to say that the generation system designer would prefer the use of chopper regulators for they produce rather a lot of electrical noise.

We should note that avionic equipment is not the only source of heat due to voltage excursions; relays and solenoids designed to work on emergency voltage levels must also work at normal voltages, or even the top limits of abnormal voltages, and in doing so dissipate considerable quantities of heat.

Table 1 shows the typical ranges of a.c. and d.c. voltages the sources of which will be explained later; of particular note is the possible range of 16 to 30,5 V on the d.c. system for Category B equipment.

Upward voltage transients are catered for by equipment regulators but downward transients require power storage. It is a rather sobering thought that certain modern avionic equipments, whose designers found power storage components too large, are designed with voltage regulators operating at the lowest (normal system operation) transient levels! Interruptions of supply or more severe transients which cannot be easily catered for by power storage are usually overcome by allowing the equipment to draw its power from an uncorrupted bus bar, often the battery bus bar, during the disturbance. However, such measures involve extra circuitry with its inherent decrease in system reliability and increase in dissipation.

The less obvious deviations from the ideal supply such as voltage modulation, voltage unbalance between phases and uneven phase displacement tend to cause problems with high voltage equipments, usually transmitters, requiring more stabilising components.

Variations in frequency do not seem to excite the same concern with designers though designing for the lowest frequency increases the sizes of transformers, fans and pump motors; perhaps more important fans and pumps may dissipate more energy than necessary when running above or below nominal design speed.

Harmonic distortion is certainly a problem and can cause havoc in rectifier systems where excessive ripple is produced on certain waveshapes which in turn demands more regulation with inherent dissipation.

Having discussed the various problems, albeit briefly and from a consumer viewpoint, which arise due to deviations from the ideal supply let us now review the components of a simple generation system to establish the sources of the deviations.

3. THE COMPONENTS OF A SIMPLE MODERN GENERATION SYSTEM

3.1 The a.c. System

The basic components of a simple a.c. generation system are as follows:-

a) the generator's constant speed drive (C.S.D.)

b) the generator

c) the generator's control and protection unit (C.P.U.)

d) the generator line contactor (G.C.)

e) bus bar protection devices such as fuses or thermal circuit breakers

f) cables from bus bar to consumer equipment.

Let us consider these items in order:

a) The C.S.D. - to generate at constant frequency the generator must revolve at constant speed but usually the engine driving the generator has a speed range of about 2:1 so a constant output speed device must be interposed between the engine and generator.

Several techniques are used to produce C.S.D.s, typical examples being:

Fig. 1 - The engine shaft feeds the input of a differential gearbox whose output drives the generator. A hydraulic reversible pump/motor arrangement of variable stroke is interposed between the input and the control leg of the differential gearbox and seves to add or subtract speed under the control of a governor driven from the output (generator) shaft thus maintaining the output speed reasonably constant.

Fig. 2 - This unit utilises a differential gearbox as Fig. 1 but speed control is achieved with an Eddy Current Brake on the control leg.

Other units use air turbines or various types of friction drives (using variable contact paths).

Common to all of these units is an output speed governor, which ideally should eliminate all output speed, and therefore frequency, variations . Inevitably, however, the governor control loop including the variable speed unit suffers from setting tolerances, temperature drift, friction, worn components, non-ideal control laws etc., which together produce the normal steady state frequency range of 380-420 Hz of Table 2 and the transient response curves shown on Fig. 3.

It is unlikely that such mechanical control systems will improve dramatically and the only solution is to use electronic frequency control. These so called variable speed constant frequency (V.S.C.F.) systems generate variable frequency power and from this source electronically synthesise the required frequency. Though producing much tighter frequency control, electronic V.S.C.F. systems suffer from; a large volume for the control unit (which also requires large coolant flow rates due to high dissipation); a larger generator than required with the mechanical C.S.D. and various other limitations. V.S.C.F. systems have not yet gained much favour and only experimental systems are currently flying.

Apart from steady state limits and transients in frequency, specifications of power supply quality usually quote values for;

(i) frequency modulation - this is the cyclic or random variation of frequency due to imperfect damping of the C.S.D. control loop and the effects of minor input speed variations; typically less than ± 4 Hz from the nominal.

(ii) frequency drift - this is gradual change of output frequency due to component wear (long term) or temperature changes (short term) the latter occurring at rates less than 15 Hz/min.

Not all C.S.D.s produce the full range of variations quoted in supply specifications but as equipment usually has to be capable of working on a variety of systems on different aircraft these advantages cannot be utilised.

b) The generator - when driven by the C.S.D. this must produce a waveform as near as possible to a sine wave. The typical machine is configured as shown in Fig. 4. Stage 1, called the pilot excitor, employs a permanent magnet rotor and generates a 1600 Hz 3-phase supply which powers the generator control and protection unit (C.P.U. - see (c)). From this supply the C.P.U. produces d.c. which is fed via a voltage regulator to the stator of stage 2 of the generator. Stage 2, the main excitor, generates 3-phase 1000 Hz on the rotor which it feeds to a full-wave rectifier bridge on the rotor. The d.c. output of this rectifier is fed along the rotor to stage 3 the main generator which is a 400 Hz, 3-phase, salient pole machine.

Various factors, such as ripple content of the d.c. output of the regulator and rotor rectifier bridge, iron circuit shape and winding configuration etc., affect the generated output waveform. Present equipment specifications quote the harmonic content as 8% of the fundamental with no single harmonic exceeding 5% with further limitations dictating maximum point deviation from the fundamental. In practice generator specifications quote 5% and 3% respectively to allow a margin for corruption of the supply by equipments drawing non sinusoidal currents.

In practice these limits allow the generation of waveshapes which to the eye and certain equipments, notably rectifier bridges, do not appear anything like a sinewave and generators with such bad waveshapes have been produced. Theoretical examination shows that actual quantities of harmonics are not as important as their phase relative to each other and the fundamental, and specifications for generators ought to be modified to reflect this. Generally speaking, generator waveshape refinements result in larger machines so a sensible compromise must be reached.

Having generated a good waveform we must ensure that it remains so. Rectifier and regulating circuits within consumer equipments cause considerable distortion of the current waveform drawn from the generator which in turn acts on the generator impedance to produce corruption of the voltage waveform. One solution is a lower impedance and consequently larger generator but this brings penalties of mass and large fault currents in the event of short circuit. Another solution is better design of consumer power supply circuits. Again a compromise must be reached but the authors believe that there is considerable scope for better design of avionic equipment in respect of current distortion.

Voltage transients are a function of both the generator and its voltage regulator within the C.P.U. As loads or faults are switched onto the generator its voltage falls or rises according to the magnitude of the current, the effect of the regulator and the magnitude of the generator impedance. One way of reducing transient excursions is to reduce the generator impedance but this means a larger machine with the penalties mentioned above. Regulator performance is discussed below in para. c.

Phase unbalance is due to assymmetric generator phase impedances and current loading and neither of these can be much improved with conventional generators/regulators though their is scope with V.S.C.F. machines where each phase is separately regulated.

c) The generator control and protection unit (C.P.U.) - the C.P.U. carries out two main functions, viz:-

(i) it controls generator voltage by regulating field current

(ii) it embodies various protection circuits to prevent generator overload, and equipment damage/ malfunction due to incorrect voltage or frequency.

N.S.S.L. - Consider Table 1, which shows that the normal steady state limits (N.S.S.L.) of voltage for the average of the three phases (which is what the regulator controls), lie within a 3V symmetrical band about the nominal phase voltage of 115V. This tolerance is a combination of setting tolerance, temperature drift and to some extent field current drive limitations and when expressed as a percentage is typical of a.c. or d.c. generators irrespective of their voltage. It is unlikely that this tolerance could be reasonably improved.

A.S.S.L. - Apart from transient effects during load switching or fault clearance, excursions beyond N.S.S.L. are due to faults in the regulator (whether voltage or frequency). The C.P.U. contains circuits to detect non-transient abnormal conditions and initiate action to remove its faulty generator channel from the bus bars. However, such circuits require a tolerance band, symmetrical or otherwise and it is these tolerances which give rise to the conditions known as abnormal steady state limits - A.S.S.L.

When a speed governor on a C.S.D. or a voltage regulator within a C.P.U. fails it may do so catastrophically or otherwise. If the failure is catastrophic then frequency or voltage will rapidly exceed the outer limit of the protection circuit tolerance band and the generator will be switched off the bus bar, the latter experiencing the fault conditions only transiently - depending on system type, a healthy generator may then be switched in. However, if the failure is of a minor nature, taking the form of excessive drift on setting, and the voltage or frequency drifts up to, but does not exceed, the actual protection circuit setting then no action is taken. In theory this condition can exist indefinitely at the extreme end of the upper or lower tolerance band thus we have abnormal, steady state, limits.

Thus as shown on Table 2 the A.S.S.L. of frequency are 370 and 430 Hz, that is the tolerance band extends by 10 Hz beyond the N.S.S.L.

Examining Table 1 shows the A.S.S.L. of the average of the 3-phase voltages to be 107,5 and 122,5 (typical values), that is the protection circuit tolerance bands can be 6V wide. Since the regulator control circuit tolerance is only 3V one could reasonably expect the protection circuit tolerances to be as good or better than 3V. In fact, they usually are, and they are set at the extremes of the allowable band; thus a typical system could have N.S.S.L. of 115 ± 1,5 V and protection circuits set at 109 ± 1,5 V and 121 ± 1,5 V giving A.S.S.L. of 107,5 and 122,5.

It would appear that there is an unnecessary 3V gap between N.S.S.L. and the protection circuit limits; the authors have discussed this fact with C.P.U. designers who, though admitting it is unnecessary, are reluctant to remove it!

Furthermore, recent editions of power supply specifications viz. BS-3G-100 and (yet to be issued) MIL-STD-704B are extending the A.S.S.L. to about 100V and 130V, respectively for under and over voltage, despite the fact that current C.P.U.s are not having problems in meeting the earlier issues of these specifications!

Another point worth noting is that few specifications give any indication of the probability of A S.S.L. occurring with the result that many equipments are unnecessarily designed to meet what must be exceedingly rare conditions and consequently dissipate excess heat.

The authors would like to see a study made into A.S.S.L. and their probability of occurrence and would welcome comments on this subject.

Voltage Transients and Modulation

On an unregulated generator the terminal voltage would vary as the load was varied due to the effect of the generator and feeder impedances. The regulator within the C.P.U. attempts to correct these changes as they occur by varying the generator field excitation and as a result controls the voltage within N.S.S.L.

The regulator and generator field circuits form a control system which cannot act instantaneously and so produces an error whilst it is settling towards the N.S.S.L.; this short term error produces voltage transients within the curves shown on Fig. 5.

Many equipment designers seem confused as to the exact form and reason for the shape of transients so we will attempt to clarify the subject briefly.

Consider the application of an extra load to the generator, removal of load has the inverse effect and will not be described. As the load is applied the generator voltage falls and the C.P.U. sensing this fall, at the point of regulation, increases the generator field drive voltage. The field current, however, limited by the field inductance takes time to rise so the system voltage is transiently depressed. A critically or overdamped control system would result in excessively long transients and so, as with the majority of control systems, the designer chooses components to give underdamping. The result is that although the system returns quickly to within N.S.S.L. the regulator will overswing at first producing an upward voltage swing; in fact there will be a series of damped upward and downward oscillations before the system returns within N.S.S.L.

There is a limit to the extent of improvement by increasing field drive because although the extent of a transient can be limited there will be a longer settling time. Many current systems have considerably better performance than indicated by the limits of Fig. 5 but, again, the constraints of equipment having to work on other supplies (in particular ground power supplies) of lower quality preclude our taking advantage of this performance. Perhaps we should improve all other supplies on which equipment may work rather than waste the higher quality aircraft supplies?

Finally, voltage modulation is caused by the cyclic drifting of the regulator at it tries to achieve an absolute steady state value and fails due to insufficient damping, and minor variations in speed and load; the result is small variations between the peak values of successive half cycles of the waveform which form a modulation envelope.

d) The system contactors - these devices give the ability to isolate generators from the bus bars and the latter from each other. At present the majority of these devices are electromechanical i.e. they use solenoids operating sets of contacts. As such these devices have relatively long operate times when compared to electronic switches; at the very best a 40 kVA generator contactor could have pick up and drop out times of 30 ms and 10 ms respectively, allowing for a full range of temperature.

Thus, whenever changeover from one supply to another occurs, whether in normal or abnormal conditions, there will be an interruption of supply for some tens of milliseconds. A typical normal case might be changeover from ground to aircraft power and an abnormal case the switching in of a healthy generator after switching out a faulty one.

Electronic switches offer much faster operate times but suffer from high series voltage drop and high dissipation, not to mention complication of drive circuitry and limited isolation in the 'OFF' state. Although improvements have been made in the series voltage drop overall device dissipations remain high and vast improvements are unlikely.

One way of improving matters is to make a hybrid device where the electronic unit makes connection during the pick-up time of the contactor, however, drop-out time is not improved.

Such devices, (the authors have experience with the d.c. variety) while providing high speed and low dissipation, are quite complicated especially in the 3-phase unit. It will be interesting to see whether industry will produce reasonably reliable and cheap hybrid contactors or whether system designers will allow sensitive equipments to incorporate electronic switches to maintain rapid switching for limited loads.

e) Bus bar protection devices - i.e. fuse links or thermal circuit breakers are installed between the bus bars and consumer equipment feeders to prevent cable fires and reduce bus bar disturbances.

Protection devices limit voltage transient extent and duration, on the bus bar in the event of consumer circuit faults, so the faster the device operates the better. Without examining fuse and circuit breaker characteristics in depth it is sufficient to say that both give adequate performance. Fuses give slightly faster operation and better current limiting than circuit breakers but the latter can be more easily matched to cables and have advantages on 3-phase circuits and circuits with high inrush currents. It is usual to find both devices in a modern power system with circuit breakers more predominant.

f) The consumer equipment feeder cable - the power system regulates voltage at the bus bars so any impedance between that point and equipment terminals causes further voltage drop aggravating the already large tolerances on N.S.S.L. or A.S.S.L.

Cables vary in length according to equipment position so feeder impedance is specified as a maximum voltage drop based on the normal R.M.S. equipment current. Because this value is expressed as a maximum equipments are designed to regulate at the lowest voltage, consequently, even if the volt drop turns out less the system dissipation does not reduce;it is merely redistributed. The actual values are specified on Table 1 where it will be seen that there are three categories; category B is most commonly specified, category A is rarely allowed and is for equipment with special voltage difficulties, category C is for equipments drawing pulsed currents.

The problem with tightening up these limits to reduce dissipation is the extra weight and volume associated with lower impedance cables; this would affect smaller aircraft less than larger since the latter will probably have more cables on the limit of voltage drop. (Although growth laws will obviously be more severe on small aircraft)

However, with system dissipations rising and affecting aircraft performance perhaps now is the time to review the situation and establish whether the present values are still optimum.

3.2 The d.c. System

Most of the tolerances of the d.c. system arise for the same reasons as those on the a.c. system described above and so will be covered in a little less detail; indeed the effects of contactors, fuses, circuit breakers and feeder cables are the same and will not be covered again.

There are two methods of obtaining our d.c. supplies; they can be generated in a dedicated d.c. generator controlled by its own C.P.U. or the supply can be obtained from a transformer rectifier unit (T.R.U.) fed from the a.c. supply.

The choice depends on the quantity of d.c. required, (assuming we have got an a.c. supply) length of feeder cables and the philosophy of redundancy particular to the aircraft in question. T.R.U. supplies are generally 28V so we will compare the performance of T.R.U.s and generators at this voltage for convenience.

A T.R.U. basically transforms all the voltage variations, both transient and steady state, presented to it by the a.c. supply. Thus on a percentage basis a T.R.U. d.c. supply can be no better than its a.c. source, in fact due to transformer winding and diode voltage drops it is much worse.

A dedicated d.c. generator on the other hand regulates its supply at the bus bar and so has voltage limits equivalent to an a.c. machine (it is of course an a.c. machine plus rectifier).

Again the need for equipments to work on more than one aircraft and hence possibly more than one supply type dictate that the widest limits be specified, as evidenced in Table 1 which covers both T.R.U. and generator N.S.S.L. and A.S.S.L.

It is worth noting that if A.S.S.L. could be ignored on the grounds of exceedingly low probability (as considered in the section on a.c. A.S.S.L.) then for some equipments the bus bar voltage tolerance could be specified as close as 28 ± 0,5 V when fed from a d.c. generator.

3.3 The Emergency Supply

Emergency supplies are not always available and even when they are they are of limited capacity and usually limited duration. Emergency generators, a.c. or d.c., whether fed from monofuel motors or air turbines can generally have similar tolerances to the normal supply and as such do not concern us. The alternative source of d.c. is a battery possibly driving an invertor to provide a.c.; this type of emergency source has much wider tolerances than the normal supply and does concern us.

As Table 1 shows the emergency d.c. bus bar voltage varies from 18V to 29V; this range in fact covers all types of emergency supplies but the lower voltages are typical of battery supplies. These low voltage battery supplies concern us because equipment designed to work on such supplies generally has high dissipation when fed with the normal supply.

The lower limit is a function of the battery's state of charge and could be raised by increasing battery capacity, for fixed emergency load and duration. However, even to raise the limit to the lower A.S.S.L. would involve using huge batteries.

Invertor supplies are kept to a minimum because invertors have low efficiency. Most invertor fed equipments, such as gyros, can perform quite adequately on a square wave supply yet most invertors go to great lengths, with low efficiency, to produce a reasonable sine wave.

The authors would be pleased to hear the readers' opinions of using square wave invertors and indeed we believe that the whole field of emergency power sources should be reviewed, not in isolation but as a total aircraft system concept.

4. GENERATION SYSTEM DESIGNS - THEIR EFFECTS ON CONSUMER EQUIPMENT

The above paragraphs have described the individual components of a simple a.c. and d.c. generation system and how their performance affects consumer equipment.

In this section we present some of the factors affecting the way in which we interconnect these components to form single or multichannel systems.

4.1 a.c., d.c. or both? - this is the first question and many designers would answer 'both' from experience, nevertheless, we will examine the choice as follows:-

4.1.1 a.c. advantages

(i) a.c. is more suitable for motor loads as it obviates brush gear or bulky high dissipation electronic commutation,

(ii) different voltage levels can easily be produced by transformers

(iii) it is relatively easy to switch at high voltage

(iv) synchro references are available.

4.1.2 a.c. disadvantages

(i) difficulty in obtaining a good waveform, and maintaining it

(ii) complicated high dissipation constant speed drives or electronics required to generate constant frequency from varying input speed

(iii) relays, solenoids, magnetic indicators, etc., are larger when designed for a.c. rather than d.c.

(iv) essential a.c. equipments require invertor drives in emergencies though the use of square wave invertors would make this less of a penalty.

4.1.3 d.c. advantages

(i) generation as frequency wild a.c. followed by rectification obviates C.S.D. techniques. Waveshape problems only appear as ripple and are not a serious problem

(ii) emergency supplies can be available direct from batteries.

4.1.4 d.c. disadvantages

(i) the absence of current zeros, typical of an a.c. waveform, makes the switching (OFF) of d.c. difficult. Low voltages (28V) are therefore commonly used though 112V d.c. system have been used.

(ii) if high voltage d.c. were used modern avionic equipment power supply stages would be more complicated and/or less efficient

(iii) generator feeder magnetic fields can only be balanced by running a dedicated negative feed with the positive producing roughly twice the feeder dissipation as a balanced 3-phase supply using airframe as neutral and twisted phase feeders to reduce magnetic fields.

4.1.5 Thus to some extent the choice is dependent on the type of load being supplied but it is also largely affected by the characteristics of existing equipment chosen on cost grounds.

4.2 a.c and d.c. Voltage Levels?

Whether we have chosen a.c., d.c. or both there remains the question of the supply voltage.

4.2.1 The a.c. system voltage? - Higher voltages lead to lower currents, for a given power, and hence smaller cable dissipations or smaller cables; however, higher voltages bring switching and insulation problems. The present 115/200V systems seem adequate for power loads, perhaps avionic equipment designers would like to comment on them?

4.2.2 The d.c. system voltage? - two voltage levels are in use, 112V and 28V. The former, though seldom used now, is more suitable for heavy power loads though it brings problems with switchgear. The 28V level is more suitable for relay coils, solenoids, magnetic indicators, etc., and eases switchgear problems, moreover, it is much nearer the requirements of transistor equipment.

4.2.3 The choice? - generally speaking high power loads suit high voltages and low power loads suit low voltages. The system choice is influenced by the proportion of high and low power loads but as these are often reasonably equally split it is common to find both high and low voltages available. An a.c. system is usually chosen for the high voltage supply, to obviate switchgear problems and motor brushgear, and the low power loads are fed from 28V d.c.

However, there has recently been a good deal of discussion centred on the possibility of providing close tolerance regulated supplies, dedicated to avionic equipment, of say ± 15V and 5V. Because of their bearing on total system integrity and dissipation these supplies are discussed in section 6.

4.3 A Single or Multichannel System?

A simple system could have a 115/200V a.c. channel and a 28V d.c. channel with perhaps a battery or some other emergency back up. However, the loss of either channel would mean that the aircraft would be on emergency power and the mission would be aborted.

It is common, therefore, to have at least two channels of each voltage level in order that single failures do not result in mission abort.

Let us consider the various configurations of multichannel a.c. and d.c. systems to see how they affect supply quality; for clarity only twin channel systems are covered, more channels only extend the basic arguments.

4.4 Configuration of a Twin Channel a.c. System

Consider a basic twin channel 115/200V a.c. system as shown in Fig. 6. The system consists of two generators G1 and G2, a ground power source G.P., the generator contactors GC1 and GC2, the bus tie contactors BTC1 and BTC2 and the bus bars BB1 and BB2.

These two channels can be operated in a number of basic modes which are described in the following paragraphs:

4.4.1 System A

Perhaps the simplest system one can envisage is with GC1 and GC2 closed and the BTCs open (reversed for ground power operation). In these conditions each channel operates in complete isolation and in the event of generator failure there is no closing of the BTCs.

4.4.1.1 System A, advantages:-

(i) Complete isolation between channels is maintained, the probability of both BTCs closing accidentally, due to signal faults etc., when either bus bar is faulted is negligible.

4.4.1.2 System A, disadvantages:-

(i) Loss of either channel means loss of all equipment on the associated bus bar which can mean loss of essential equipment, unless there is duplication of the latter.

(ii) There is no frequency or phase locking of the channels. System design can obviate the need for phase locking and careful looming can reduce beat frequency problems due to the lack of frequency locking. Frequency locking can be provided but defeats the object of complete isolation.

(iii) The full capacity of the system (i.e. both generators) is not utilised to minimise transient levels and durations.

4.4.2 System B

This system has only one bus bar, i.e. the BTCs on Fig. 6 are replaced by a permanent link and a separate ground power contactor. Only one generator is connected to the bus bar, the other remains on standby to be switched in if the first generator fails and is switched off.

4.4.2.1 System B, advantages:-

(i) Loss of one generator does not result in loss of any equipment.

(ii) Supply from one generator therefore no problems due to phase and frequency locking.

(iii) Simple bus bar system.

4.4.2.2 System B, disadvantages:-

(i) In the event, albeit unlikely, of bus bar short circuit all equipment supplies are lost.

(ii) In the event of generator failure there is a complete interruption of power to equipment as one GC opens and the other closes.

(iii) The full system capacity cannot be used to minimise transient levels and durations.

4.4.3 System C

This is similar to system A but in the event of a generator failure the BTCs close allowing the remaining generator to feed both bus bars.

4.4.3.1 System C, advantages:-

(i) Loss of one generator does not cause loss of any equipment and only one bus bar sees an interruption of power.

4.4.3.2 System C, disadvantages:-

(i) Overcurrent protection system required on bus bar link to prevent cascade faults.

(ii) No phase or frequency locking though the latter can be catered for with C.S.D. governor control circuitry.

(iii) The full capacity of the system is not utilised to limit transient levels and durations.

(iv) In the event of generator failure one bus bar sees an interruption of supply.

4.4.4 System D

This is a parallel system i.e. the BTCs and GCs of Fig. 6 are all closed in normal operation (a separate GP contactor will be fitted and only one BTC is essential).

4.4.4.1 System D, advantages:-

(i) Paralleling the generators gives inherent phase and frequency locking.

(ii) The combined capacity of two generators minimises transient levels.

(iii) Generator failures need only cause a transient, though this advantage is not realised by many parallel systems which open the BTC to decide which generator is at fault causing a subsequent interruption of supply.

(iv) The load sharing circuitry effectively limits the extent of A.S.S.L. when the system is paralleled.

4.4.4.2 System D, disadvantages:-

(i) A single failure of the BTC or its control circuit results in a non-paralleled system - considerable duplication is necessary to avoid this and this increases the probability of the BTC not opening in the event of a bus bar short circuit, with consequent loss of both channels.

(ii) Advantage (ii) causes problems of switchgear rupture capacity.

(iii) Control and protection circuitry is much more complicated, which raises the probability of incorrect action causing cascade failures.

4.4.5 System A, B, C or D?

Consider first the non-paralleled systems: system A rarely gives sufficient redundancy; systems B and C are much better than A in this respect and when considering bus bar faults, which tends to be necessary on modern fly-by-wire aircraft, system C scores over B. Of these three none utilise the full system capacity to minimise transient levels and durations and only B provides freedom from phase and frequency locking problems, furthermore, all are subject to an interruption of supply on one or all bus bars in the event of a single generator failure.

System D, the paralleled system, gives adequate redundancy of supply and bus bars, can eliminate in flight interruptions of supply on all bus bars (though they still occur during changeover to and from ground power), has inherent phase and frequency locking and minimises transient limits and durations. However, unless the bus bar link circuit incorporates reasonable redundancy with its increased risk of cascade failures, the phase and frequency locking cannot be guaranteed. Also, in the event of one generator failure all equipments will see transient levels typical of a single generator capacity bus bar.

Thus, inevitably the choice is dependent on many factors peculiar to each individual aircraft system philosophy.

However, the authors would like to hear readers' views on the necessity of providing either phase or frequency locked supplies.

4.5 Configuration of a Twin Channel d.c. System

Whether fed from d.c. generators or transformer rectifier units, at high or low voltage, the d.c. system can be arranged with similar combinations to its a.c. counterpart.

The arguments for and against the systems are similar though phase and frequency locking are not relevant. Paralleling supplies can be simpler and overcurrent protection on bus bar links can be as simple as fuse links.

One point worth noting is that if a battery is fitted to the essential bus bar, directly or via a high speed hybrid d.c. contactor, then its capacity can be used to minimise transient limits and durations on that bus bar and effectively eliminate interruptions (thus a type B system can be made more attractive than its a.c. equivalent).

Again, the choice of system depends on the individual aircraft requirements for quality and redundancy.

5. THE OVERALL COOLING PROBLEM

So far, this paper has been analysing the sources of power supply tolerance that eventually lead to equipment dissipation with the aim of minimising the latter. For completeness this section looks briefly at several areas where the generation equipment dissipation could perhaps be reduced.

(i) Power for heavy loads such as intake heaters, E.C.M. pods and fuel pumps, which are predominantly in the centre of the aircraft, often comes from bus bars located in the front of the aircraft with consequent excessive cable dissipation and mass. The effects, on total cable dissipation, of siting bus bars more centrally on the aircraft should be reviewed and consideration given to feeding some of the heavy loads upstream (i.e. nearer the generators) of the main bus bar point of voltage regulation.

(ii) Generator and T.R.U. efficiencies are as high as one could reasonably expect for the size of unit currently used but C.S.D.s or their electronic equivalents have quite high dissipations. It could be useful, therefore, to critically examine the need for generating at constant frequency, the overall system dissipation may be less with variable frequency.

(iii) More effort should be directed towards using waste heat to replace electrical heaters which indirectly only generate more waste heat. The need for electrically driven fuel pumps should be critically examined, many aircraft manage using direct engine power to drive the pumps via mechanical or fueldraulic links.

The point being made is that heat dissipation is not an equipment problem it is a system problem.

6. NEW CONCEPTS - THE CENTRALISED POWER UNIT

The term Centralised Power Unit means different things to various people but generally covers the concept of a T.R.U. with one or more closely regulated outputs at voltage levels more suited to the requirements of present day electronics (e.g. +5V, ±15V etc.). This concept aims to reduce system losses by centralising the production and regulation of electronic voltage levels enabling larger more efficient units to be used.

We now examine the feasibility of the concept:

The basic concept of several voltage levels is reasonable, though their number would have to be very limited so considerable standardisation would be required within electronic units. Moreover, regulators of some form would still be required in electronic equipments to remove the effect of indeterminate line volt drops. Very low voltage levels e.g. 5V for T.T.L. would not really be feasible since line losses (voltage and dissipation) would be too great using reasonable cable sizes.

Thus, so far, we have established that we can only go part way towards the ideal, now let us examine the problem of tightening up on tolerances.

Earlier in this paper we considered the sources of voltage tolerance and it was shown that the basic T.R.U. can achieve 26-29V in N.S.S.L. due to N.S.S.L. a.c. inputs plus its own impedance or 24,5-30,5V with A.S.S.L. inputs.

A regulator in the T.R.U. could be set at 24 ± 0,5V and allowing 2V line drop would give a range of 21,5-24,5 at the equipment terminals for an input range of 24,5-30,5V.

However, such a regulator could fail therefore an overvoltage protection unit would be required to remove the faulty T.R.U. This inevitably leads to the possibility of an upper A.S.S.L. of say 26V if the protection were set at 25,5 ± 0,5V. Thus equipment would be required to work over a terminal voltage range of 21,5-26V compared with the previous 22,5-30,5 (we have ignored emergency conditions).

At this stage the improvement does not look as though it will be very worth while, especially since the degradation of T.R.U. reliability will demand extra T.R.U.s whose consequent lighter loading will give lower overall efficiency.

However, if we can ignore the probability of A.S.S.L. occurring at input or output a 3V spread at equipment terminals becomes feasible i.e. 1V for regulator tolerance and 2V line drop. This is of course what a good d.c. generator can give already within N.S.S.L.!

Without labouring the point further we have tried to show the limitations of the Centralised Power Unit concept though a detailed study is required to see just how useful this concept could be.

7. CONCLUSIONS

This paper has examined generation system components and their deployment to form systems with a view to examining the sources of the various voltage and frequency deviations which bear on equipment performance and dissipation.

It is perhaps not surprising that no obvious sources of improvement were highlighted in the generation equipment itself.

The authors feel that several areas, such as the use of Centralised Power Units, reduction of electrical heaters and fuel pumps, use of wild frequency a.c. etc., warrant further study but that the greatest savings would possibly result from philosophy changes such as

a) Design all equipment to give full performance on N.S.S.L. only and possibly even discount A.S.S.L. altogether - a study is required to determine the probability of A.S.S.L.

b) Do not rely on batteries for emergency d.c. supplies - in conjunction with (a) above this means equipment need only work over N.S.S.L. plus line voltage drops. A study on the penalties of emergency d.c. generators driven by airturbines or monofuel motors is required.

Overriding all other factors today is probably total system cost, so whatever we do to improve matters it must be amenable to standardisation over a range of aircraft.

Finally, the authors would like to thank the Directors of the British Aircraft Corporation Limited for their permission to publish this paper. Any opinions are those of the authors and do not necessarily reflect the views of B.A.C.

System Conditions	a.c. System r.m.s. Voltage at Point of Regulation (Volts)		d.c. System Voltage at Point of Regulation (Volts)
	Average of 3-Phases	Individual Phase	
Normal Steady State Limits (N.S.S.L.)	113,5 - 116,5	112 - 118	26 - 29
Abnormal Steady State Limits (A.S.S.L.)	107,5 - 122,5	106 - 124	24,5 - 30,5
Emergency Limits	110 - 120	108 - 122	18 - 29
Equipment Category	**r.m.s. Volt Drop From Point of Regulation to Equipment Terminals**		**Volt Drop From Point of Regulation to Equipment Terminals**
A	2		1
B	4		2
C	8		3

TABLE 1 - TYPICAL POWER SUPPLY VOLTAGE RANGES

System Conditions	System Frequency (Hz)
Normal Steady State Limits (N.S.S.L.)	380 - 420
Abnormal Steady State Limits (A.S.S.L.)	370 - 430
Emergency Limits	360 - 440

TABLE 2 - TYPICAL POWER SUPPLY FREQUENCY RANGES

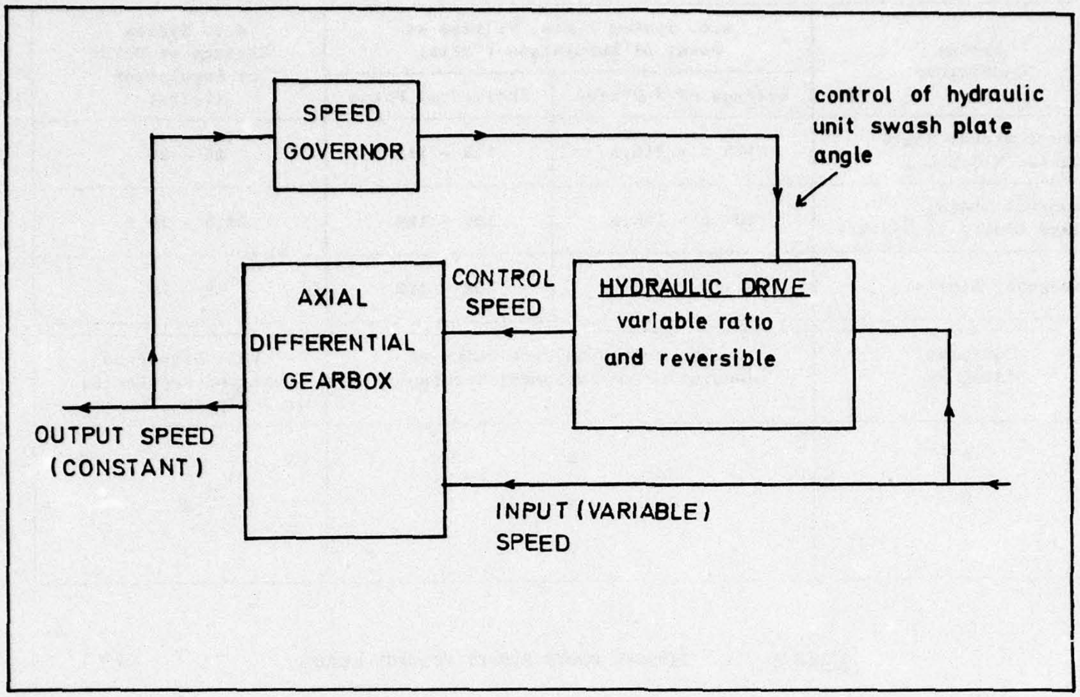


FIG. 1. SCHEMATIC OF HYDROMECHANICAL CONSTANT SPEED DRIVE.

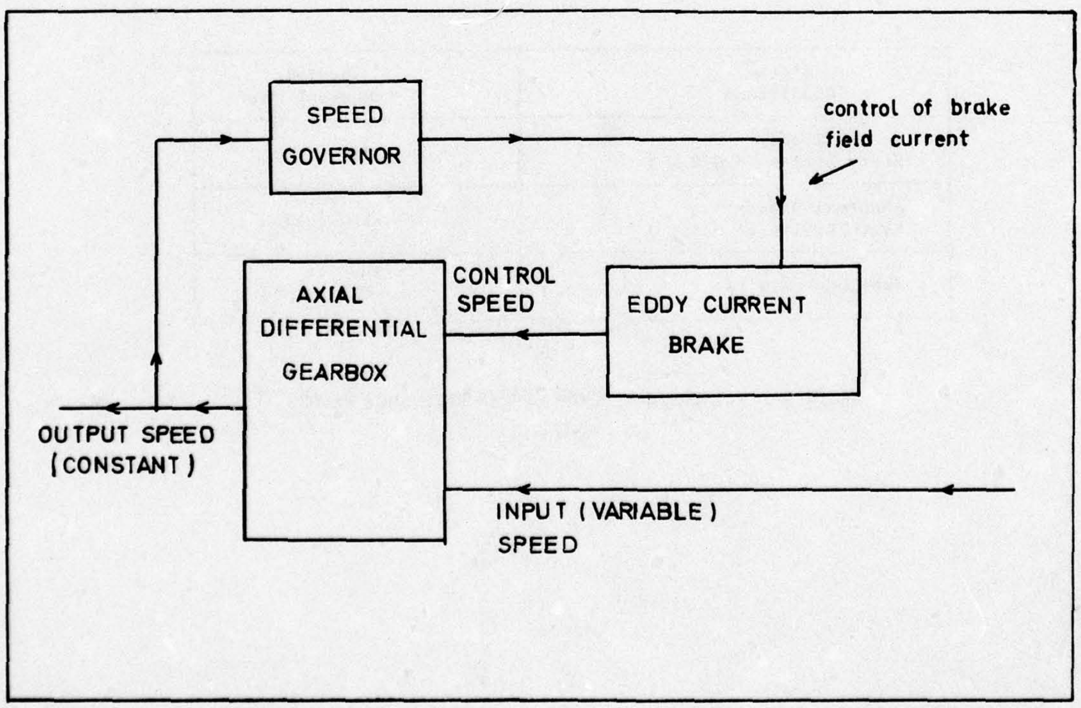
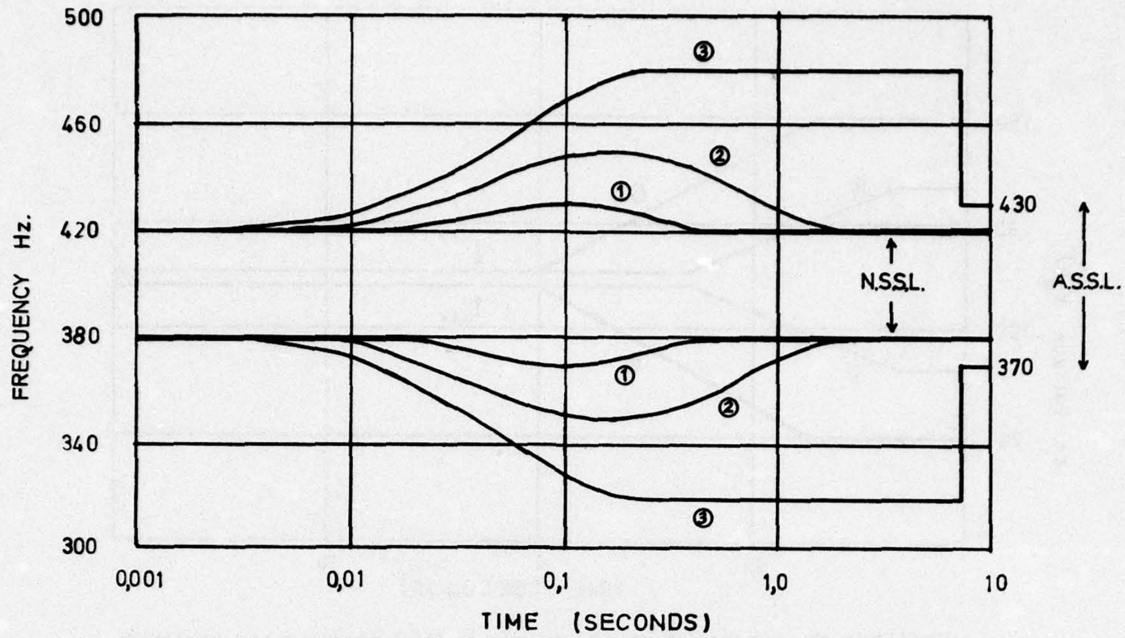


FIG. 2. SCHEMATIC OF EDDY CURRENT BRAKE CONSTANT SPEED DRIVE.



LIMIT 1 INCLUDES THE EFFECTS OF LOAD SWITCHES ≯ 10% to 85% or 85% to 10% of FULL LOAD.
LIMIT 2 INCLUDES THE EFFECTS OF LOAD SWITCHES ≯ 20% to 170% or 170% to 20% of FULL LOAD, or MAXIMUM ACCELERATION.
LIMIT 3 INCLUDES THE EFFECTS OF FAULT LEVEL LOADS.

FIG. 3 TRANSIENT FREQUENCY LIMITS.

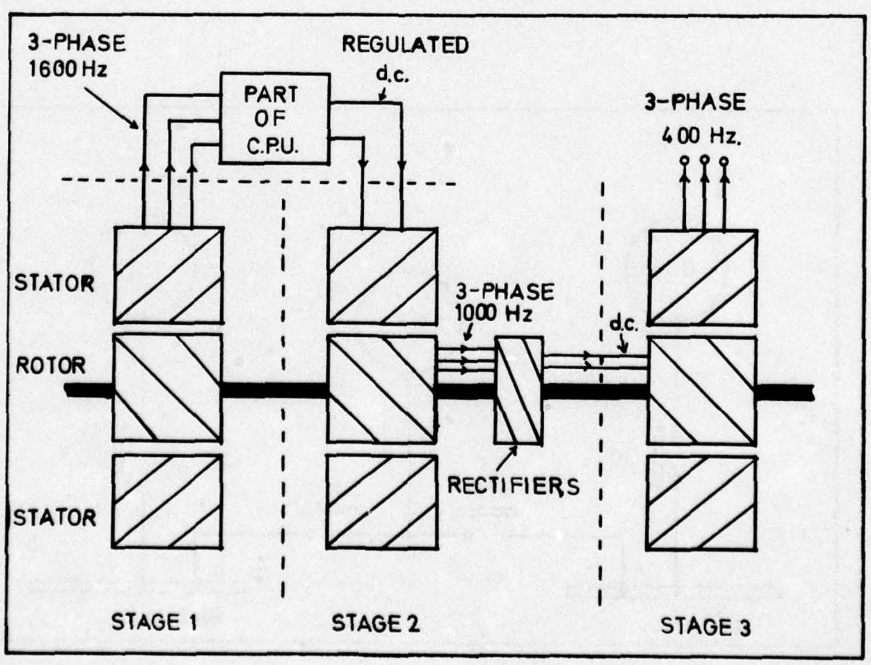
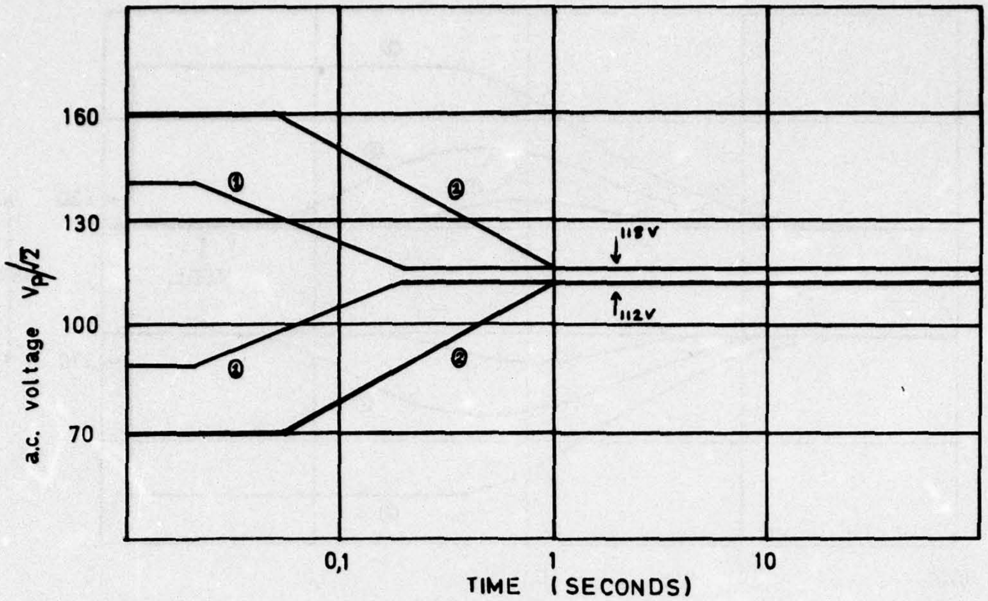


FIG. 4 SCHEMATIC 3-PHASE BRUSHLESS GENERATOR.



limit 1 is the envelope of transients due to load switches not exceeding 10 % to 85 % or 85 % to 10 % of full load.

limit 2 is the envelope of transients due to load switches not exceeding 20 % to 170 % or 170 % to 20 % of full load.

FIG 5 a.c. VOLTAGE TRANSIENTS

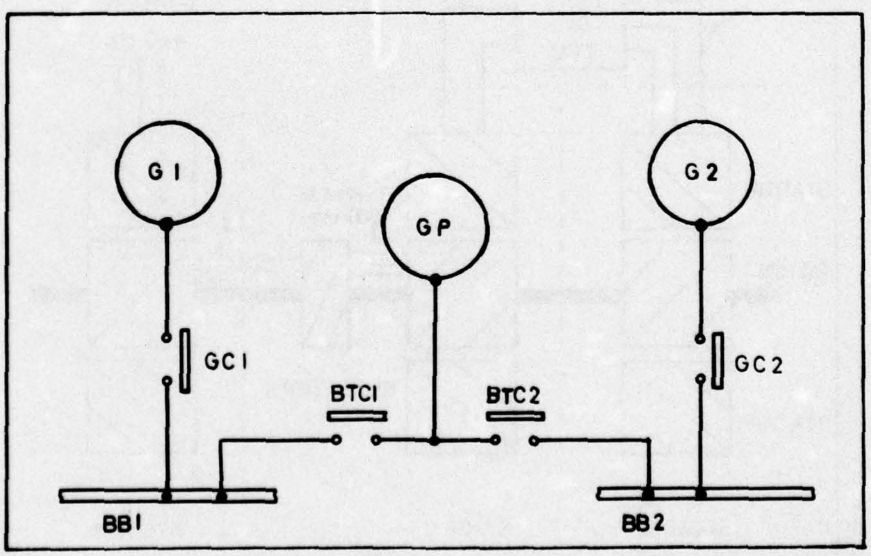


FIG 6 SCHEMATIC TWIN CHANNEL GENERATION SYSTEM

DISCUSSION

G Brogonovo:

In your Paper you have mentioned a low density of batteries as a power source. Could they be used as an emergency power supply for a light aircraft, (under 10,000 lbs weight)?

K P Gerrity and R F Bertolini:

The choice of emergency supplies must be approached as a total system problem, and depends on a particular aircraft in question. I do not feel qualified to suggest any alternative on an aircraft in this mass class though possibly the higher energy density of primary rather than secondary batteries could be a reasonable solution.

A Templeton:

While I agree that there are many wave forms which are worse than a pure sine wave, there could be some wave forms which are better. Before a choice limit is placed on harmonic content, would it be better to investigate the effects of non sinusoidal wave forms on equipment and generator and what gains could be achieved?

K P Gerrity and R F Bertolini:

While we accept that sinusoidal wave forms are not necessarily optimum, indeed in the Paper we propose questions on the use of square waves output inverters, we are limited by the considerable investment of capital over the last few decades in support of equipment based on sine wave supplies.

GENERATIONS ELECTRIQUES ET RESEAUX DE BORD
DANS LES AVIONS MODERNES

par

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RESUME

Cet exposé présente le point de vue d'un constructeur d'avions pour faire face aux problèmes posés, à bord des avions récents, par la compatibilité entre les systèmes d'alimentation et les équipements consommateurs. Il est arrivé qu'on ait favorisé les uns au détriment des autres, alors qu'une étude plus approfondie des données aurait permis une meilleure adaptation de la puissance, de la place et du poids.

Après un bref rappel des différents types de réseaux rencontrés à bord des aéronefs, l'auteur indique, pour les différentes sources d'énergie électrique, les limitations de caractéristiques qui intéressent directement les équipements utilisateurs. On examine ensuite les différentes "Zones de fonctionnement" (normales, rares, exceptionnelles) et types de "régimes" (permanents ou transitoires) applicables aux alimentations et utilisateurs, de façon plus appropriée que les exigences des normes nationales actuelles. En conséquence, une meilleure approche des définitions de chaque cas permettrait des gains appréciables.

1. GENERALITES SUR L'ALIMENTATION ELECTRIQUE.

Le fonctionnement des avions modernes repose essentiellement sur leurs installations électriques qui forment des systèmes plus ou moins complexes. Chacun de ceux-ci constitue un réseau comportant tous les générateurs, lignes, appareillages et équipements connectés entre eux. Un aéronef peut disposer de plusieurs réseaux, chacun pouvant avoir plusieurs sources de puissance.

La définition de ces réseaux est établie depuis longtemps. Nous distinguerons :

Les réseaux à courant CONTINU, dont la tension nominale est de 28 volts.

Ils fonctionnent dans une assez large plage de tension, puisqu'ils doivent assurer la recharge la plus complète possible des batteries et rester utilisables jusqu'à la fin de la décharge de celles-ci. En outre, les systèmes de régulation ou de redressement qui les alimentent y introduisent fatalement des ondulations dont le spectre de fréquence est très large (jusque vers 100 kHertz).

Malgré cet inconvénient et l'importance des câbles qui distribuent la puissance "continue", ces réseaux sont et resteront très utilisés. En effet, la tension basse nécessite peu de précautions (mis à part certaines commutations selfiques) ; les batteries qui les alimentent en secours répondent à des standards d'encombrement et de sécurité ; les basses tensions continues sont énormément utilisées en électromécanique et en électronique ; Les appareillages de coupure grossiront obligatoirement quand on augmentera cette tension, sans remplacer totalement le besoin d'une basse tension continue.

Les réseaux à courant ALTERNATIF.

A l'origine, leur fréquence variait dans une plage relativement peu étendue autour de la valeur de 400 Hz, qui avait été choisie comme le meilleur compromis entre volume, poids, prix et possibilité de réaliser des tôles magnétiques convenables.

- Les besoins des équipements et l'évolution des techniques ont conduit à "fixer" cette fréquence avec une faible tolérance, d'où la généralisation des réseaux "alternatif" à :

400 Hz - 115 V (tension simple) - 200 V (entre phases).

Beaucoup de petits consommateurs restent monophasés, alors que les gros (même en triphasé) posent des problèmes d'équilibrage des phases. La distribution de cette énergie est très demandée : abaissement ou élévation facile de la tension, utilisation de moteurs asynchrones à vitesse assez précise, fréquence de référence fixe pour calculs divers, etc...

Ce type d'alimentation est tellement généralisé qu'on l'utilise même pour des fonctions où elle devient très coûteuse. En effet, la régulation de la vitesse des alternateurs abaisse énormément leur rendement (40 % pour les petites puissances, 60 % pour les plus grosses). Or, il est moins difficile aujourd'hui de réduire les masses et encombrements de ces générateurs que de remonter leurs rendements, ce qui pose un problème pour les avions à hautes performances.

Le mauvais rendement de la génération à 400 Hz est dû principalement à la plus ou moins large plage de vitesses d'entraînement des générateurs qu'il faut transformer en vitesse fixe régulée.

- Pour réduire la quantité de chaleur à évacuer, plus encore que pour alléger les transmissions de puissance, certains avions possèdent des alimentations de puissance à fréquence variable. Ce type d'alimentation, normalisé jusqu'à présent de 350 à 600 Hz, est utilisé par des équipements de servitude (ex : Avion BREGUET-DASSAULT "ATLANTIQUE").

Des études sont en cours en vue de porter, si possible, la plage de fréquence jusqu'au-delà de 1000 Hz. Cette puissance aurait une utilisation très générale, exemple : tous les consommateurs qui redressent le courant alternatif. Certains moteurs seraient alourdis, exemple : électropompes à 2 régimes de vitesses ou doublées en nombre aux basses fréquences ; d'autres moteurs de faible puissance verraient leur rendement diminuer, exemple : ventilateurs.

Par contre, les équipements de précision exigeront toujours une alimentation à fréquence fixe, exemples : gyroscopes, machines synchro. Quoi qu'il en soit, les gains de poids sur les générateurs et sur les dispositifs de refroidissement seront importants.

Des alimentations spéciales existent sur certains avions pour résoudre des problèmes particuliers :

- Réseaux alternatifs à tension élevée (ex : 230/400 volts), limitée par les seuils d'isolement des composants, pour gagner sur le poids du câblage.
- Circuits localisés à courant alternatif redressé ou à forme d'onde non sinusoidale pour des petits utilisateurs dont la fiabilité est liée à l'indépendance ou à la redondance.
- Chargeurs à tension programmable pour batteries spécialisées.
- Circuits alternatifs 26 volts/400 Hz mono ou biphasés, utilisés dans les systèmes de calcul.
- Circuits basse tension d'éclairage ou de signalisation (5 volts en général).
- Transmission de signaux digitaux par lignes spéciales.
- L'augmentation de la tension des réseaux "continus" ferait gagner sur le poids de cuivre à installer, mais augmenterait vite le poids et le volume de l'appareillage (isolement, coupure, commutation), sans supprimer le "continu" basse tension indispensable.

2. GENERATEURS DE PUISSANCE.

Les améliorations constamment apportées aux sources de puissance électrique ne peuvent faire l'objet de cet exposé, mais certaines différences et les tendances de leur évolution permettent de les caractériser ici.

2.1. Les batteries d'accumulateurs.

Malgré les impératifs d'entretien des batteries et le soin à apporter à leurs conditions de recharge, celles-ci restent la seule réserve d'énergie électrique immédiatement disponible dans les cas critiques. Le type le plus généralisé est la batterie alcaline au Nickel-Cadmium, directement ou non raccordée au réseau 28 volts.

- Leur capacité déterminée par les besoins prévus (démarrage autonome des moteurs de l'avion ou durée de vol en "détresse") peut aller de quelques ampères-heure à 45 Ah. Cette capacité est définie par le courant constant qu'elle peut débiter pendant 1 heure dans des conditions normales de température (20 à 25°C) avant de tomber à 20 volts à ses bornes. Il faut se méfier des capacités données pour des durées de décharge plus longues.

- Les courants de recharge varient avec la tension appliquée aux bornes, avec l'état initial et la durée de la recharge, avec le nombre d'élements et avec la température de ceux-ci. On voit apparaître des dispositifs de protection ou des chargeurs plus ou moins complexes, destinés à intervenir en cas de danger par surcharge ou à empêcher ces cas. En principe, chaque batterie est seule en paralléle avec la (ou les) autres sources de son réseau continu, pendant toute la durée du vol, ce qui la rend immédiatement disponible (pas de coupures) et lui permet de jouer un rôle de filtre grâce à sa basse impédance (compensation des transitoires dans les 2 sens).

Avec les performances augmentées des avions (durée de vol, températures des soutes) la tendance est de séparer les batteries des réseaux d'utilisation afin d'adapter les tensions de recharge sans influencer les autres consommateurs. On rencontre ainsi, soit des batteries couplées à un chargeur et à un petit nombre d'équipements devant être utilisés en "détresse", soit des batteries qui seront isolées du réseau quand leur charge est terminée.

Il en résulte, sur le réseau, des difficultés dues aux "ronflettes" (Ripple) et aux différents transitoires. Un compromis ou le choix entre les 3 formules de recharge reste à établir, suivant les types d'aéronefs ou suivant les moyens de maintenance.

- Les limites de tension d'alimentation des équipements dépendent donc du schéma retenu :

  Faibles écarts de tension stabilisée et phénoménes transitoires plus importants et systématiques quand la batterie n'est pas en "tampon".

  Tensions stabilisées pouvant s'approcher de 30 volts et peu de transitoires (Surges) de toutes façons limités, si la batterie reste en paralléle avec les consommateurs.

2.2. Génératrices à courant continu.

Les avions, ne justifiant pas le besoin d'un générateur de courant alternatif, ont toujours une machine tournante à collecteur fournissant du courant continu, comme source primaire d'énergie pouvant aller jusqu'à 12 kW. Quand les moteurs de ces avions n'ont pas besoin d'un système de démarrage indirect (ex. turbine), une Génératrice-démarreur est adaptée aux caractéristiques de démarrage et de vitesse du moteur.

Un régulateur électronique maintient chaque fois la tension fournie au réseau dans une tolérance favorable à la recharge de la batterie qui a servi au démarrage. Malheureusement, la régulation par transistors génère une ondulation résiduelle à fréquence variable élevée plus ou moins bien filtrée. La présence de consommateurs non linéaires (Convertisseurs statiques) sur le réseau augmente ce phénoméne et peut le rendre aussi gênant qu'avec une alimentation par Transformateur-Redresseur.

En cas de Géné-Démarreur, la puissance mise en jeu pendant le lancement du moteur provoque une sous-tension importante que doivent encaisser les équipements déjà branchés sur le réseau (au sol comme en vol).

C'est pourquoi il n'y a jamais de protection de sous-tension sur ces réseaux. Par contre, il est plus facile d'y adapter une protection de surtension de génération, tarée assez bas en régime stabilisé (à partir de 30,5 volts).

Couplages.

Les génératrices à courant continu sont généralement couplées par 2 en paralléle quand il y en a plusieurs. Leur régulateur est alors prévu pour assurer un bon équilibre de leurs charges. Dans la pratique, les systèmes de régulation ne peuvent qu'équilibrer la tension ou le courant moyen, mais non en valeur de crête, d'où une augmentation sensible des ondulations résiduelles qui peuvent présenter elles-mêmes une modulation importante. C'est le cas s'il y a une troisiéme génératrice ou des consommateurs non linéaires sur le réseau.

Bien que la batterie en tampon atténue ces phénomènes, ces derniers équipements devront être convenablement filtrés à l'entrée sous peine de perturber le réseau en y introduisant leur propre résonnance.

2.3. Générateurs de courant Alternatif à fréquence constante.

Ils sont presque toujours constitués d'un alternateur triphasé tournant à vitesse constante, généralement auto-amorçable et indépendant du réseau continu de bord.

Leur entraînement est assuré par divers systémes indépendants ou intégrés avec l'alternateur ; on distingue :

- Les systémes électro-mécaniques décomposant le domaine des vitesses en plusieurs plages, où un accouplement à courant de Foucault régule la vitesse par l'intermédiaire d'un différentiel. Ce système est utilisable pour de petites et moyennes puissances (jusqu'à 25 ou 30 kVA), il a malheureusement un très mauvais rendement en fin de chaque plage. Le régulateur est commun pour la tension et la fréquence, qui sont soumises ensemble à certains transitoires de fonctionnement normal.

- Les systèmes hydro-mécaniques dans lequels un ensemble pompe-moteur hydraulique agit sur le différentiel qui entraîne l'alternateur. Ils sont utilisables pour les grosses puissances (40 à 120 kVA) mais dans des domaines de vitesse qui ne peuvent guère dépasser le rapport 2. Leur rendement est d'autant meilleur que la gamme de vitesses est réduite.

- Des systèmes uniquement hydrauliques ou mécaniques pour des petites puissances, mais dont la plage de vitesses, le rendement ou la fiabilité insuffisants limitent les applications.

- Il faut mentionner les systèmes à vitesse variable et fréquence constante (V.S.C.F.) comportant un alternateur polyphasé à fréquences variables élevées et un convertisseur de fréquence statique. Les thyristors de commutation de ce dernier reconstituent une source triphasée à 400 Hz.

Cette solution électronique nécessite de grandes précautions d'antiparasitage et de filtrage qui l'alourdissent et ne peuvent empêcher des modulations génantes pour le réseau. L'absence d'élément amortisseur, comme dans les systèmes décrits plus haut, peut faire apparaître des résonnances prohibitives dans la chaîne (depuis la prise de mouvement jusqu'aux divers consommateurs).

Les rendements insuffisants pouvant tomber à 40 % en régime croisiére, encore acceptables sur des avions à performances limitées (puissance installée et vitesse de vol), deviennent critiques sur les avions d'armes modernes. En effet, le refroidissement des générateurs ne peut se faire avec de l'air chaud à grande vitesse, ni par l'intermédiaire du pétrole consommé (plus froid) aux basses vitesses, d'où la nécessité d'installations complexes.

Les sources d'énergie primaires exigeront dans certaines configurations d'évacuer au moins 50 kW, sans compter le refroidissement des sources secondaires et des équipements consommateurs.

La régulation des générateurs, assurée par des circuits électroniques groupés ou non pour des raisons pratiques ou d'environnement, maintient tensions et fréquence dans les tolérances contractuelles, en principe meilleures que celles des normes générales des réseaux de bord (voir plus loin). La protection des réseaux contre les défaillances des générateurs est assurée par des circuits différenciés intégrés ou non avec les régulateurs.

- L'obligation de ne pas déclencher intempestivement, lorsque les paramètres atteignent les valeurs limites prévues, a conduit à décaler ou temporiser les seuils des protections. L'éventualité de dépassement momentané des limites actuelles en régime stabilisé reste cependant brève et faible.

C'est ainsi que la tension simple ne doit jamais dépasser 124 volts stabilisés et qu'il a fallu fixer les seuils de protections à 132 volts (phase la plus haute). L'expérience a confirmé le réalisme de cette marge de sécurité. Inversement pour les fréquences, les tolérances des protections peuvent être plus resserées que celles des normes, pour certaines applications.

- De même, pour éviter des transitoires de tension inutiles en cas de sous-fréquence, il est bon que ces 2 paramètres ne soient pas liés ; il peut donc arriver, en cas d'arrêt rapide des parties tournantes (anomalie), une baisse brutale de la fréquence avant que le déclenchement par la protection ne soit obtenu. Pendant ce temps (moins de 100 milli-secondes) les équipements peuvent se trouver entre 350 et 300 Hz avec une tension presque normale.

2.4. Alternateurs à fréquence variable.

Comme il a été indiqué au début de cet exposé, fournir une puissance électrique à fréquence fixe à tous les équipements est très coûteux en puissance installée. L'existence de réseaux à fréquence variable (plages larges ou étroites) et l'analyse du fonctionnement des équipements montrent qu'une grande majorité d'entre eux seraient aptes ou adaptables à une fréquence variant dans le rapport de 1 à 2,8 ou 3 (soit environ 360 à 1080 Hz). Le gain de poids (20 à 35 %) obtenu par la suppression des entraînements et la réduction des systémes de refroidissement, peut être reperdu sur l'augmentation de certains équipements (Pompes, Radar, etc...). Par contre, compte-tenu de l'augmentation d'échauffement de certains consommateurs, on peut compter sur une diminution de chaleur à évacuer de l'ordre de 50 %.

La fiabilité du système de génération sera largement augmentée, son encombrement et son entretien seront diminués. Des études sont à entreprendre pour connaître les répercussions de cette fréquence, dite "Sauvage", sur l'adaptation de certains moteurs et des gros équipements électroniques. Pour plus de détails, voir tableau n° 1.

2.5. Transformateurs - Redresseurs.

Ce sont des sources secondaires de courant redressé et filtré, alimentées par les réseaux à courant alternatif régulés en tension. Leur caractéristique de tension en charge U (I) est une droite descendante due au transformateur. Le point de fonctionnement sur cette droite est déterminé par la charge du réseau alimenté.

Quand on ne peut connaître à l'avance la zone moyenne de charge, le primaire du transformateur est équipé de prises de réglages s'échelonnant sur 1 ou 2 volts (à la sortie) pour permettre de choisir le meilleur compromis sur les tensions à faible et à forte charge. En outre, aux faibles charges, il y a une remontée de la tension bien au dessus de la droite mentionnée plus haut, due à la moindre chute de tension dans les diodes et au filtrage moins efficaces. Si on doit envisager le fonctionnement aux très faibles charges, on peut avoir jusqu'à 4 volts d'écart entre tensions à vide et à pleine charge.

La forme d'onde de la tension fournie est liée à la qualité du filtrage, lequel est plus facile avec un schéma de redressement dodécaphasé. Cette formule est à rechercher systématiquement chaque fois que la puissance à redresser est une fraction appréciable de la puissance alternative installée. Même à l'intérieur d'équipements (ex. Radar) le supplément de poids, par rapport à un redressement plus simple, est vite rattrapé. En effet, les ondulations résiduelles et les réinjections côté alternatif sont plus réduites,ce qui est d'autant plus avantageux quand il faut isoler la batterie.

On réalise actuellement des Transfo-Redresseurs qui, bien que non régulés, éliminent presque complètement la surtension aux faibles charges et réduisent de moitié l'ondulation résiduelle. Celle-ci reste cependant liée à l'importance des consommateurs de courant non linéaire sur le réseau.

2.6. Convertisseurs "Continu-Alternatif".

Ce sont des sources secondaires de courant alternatif utilisées soit sur des aéronefs n'ayant pas de source primaire d'alternatif, soit pour des réseaux privilégiés (pas de perturbations) ou de secours (détresse sur batterie). Seuls les convertisseurs statiques nous intéressent dans cet exposé.

Il est plus facile de construire des convertisseurs statiques monophasés que triphasés surtout quand la puissance dépasse quelques centaines de Volt-Ampères. Ceci pose à l'avionneur quelques problèmes du fait que sur les réseaux alternatifs il y a intérêt à avoir le moins possible de puissance non équilibrée. On fait appel à des compromis divers en multipliant ce type d'alimentation (convertisseurs proprement dit ou alimentation intégrée à certains équipements), ce qui multiplie en même temps leurs interférences sur le réseau qui les alimente.

Les rendements des convertisseurs ne sont jamais très élevés (50 à 65 %), pour cette raison également leur nombre est à réduire. Ce faisant, on facilitera en outre leurs systèmes de protection : en amont, moins de distribution, en aval, plus de puissance disponible. En effet, la puissance de surcharge des convertisseurs statiques est toujours faible, ce qui pose des problèmes de sélectivité (calibrage des protections aval).

- Les limites d'utilisation des convertisseurs dépendent des besoins qu'ils doivent couvrir. La fréquence est toujours très stable (oscillateur), la tension ne présente que peu de transitoires, toujours plus réduits qu'avec les machines tournantes ; la mise en route est extrêmement rapide.

Pour des réseaux de "Secours", le meilleur rendement devra se placer aux faibles tensions d'alimentation afin de prolonger l'autonomie sur batterie. A ce moment, on pourra tolérer de moindres performances de régulation (voir normes).

Dans tous les cas, le filtrage est très important afin de ne pas réinjecter dans le réseau continu des composantes alternatives dont les fréquences de base légèrement différentes provoquent des modulations d'ondulation résiduelle rapidement prohibitives.

3. CARACTERISTIQUES ET EXIGENCES DES SYSTEMES ELECTRIQUES.

3.1. Réseaux de bord.

Normes existantes.

Les limites des différents paramètres de l'alimentation électrique et de son utilisation sont définies par des normes nationales, assez voisines les unes des autres :
AIR - 2021 - D (révision en cours)
B.S.I.-3 G 100 : Part 3 (et I.S.O. : 1540-2)
MIL-STD-704 A (Edition B en discussion).
sont les plus connues.

Leur conception remonte à une dizaine d'années, époque à laquelle on n'avait pas encore l'expérience des calculateurs embarqués et où les divers émetteurs n'étaient pas aussi nombreux et puissants. Elles restent cependant des bases très valables. Bien des difficultés seraient évitées si réseaux et équipements leur étaient conformes, ainsi qu'aux normes d'antiparasitage électromagnétique (dont nous ne parlerons pas ici).

Excepté la plus récente des normes citées (B.S.I.), aucune ne donne de méthodes de vérification de compatibilité. Toutes sont insuffisantes en ce qui concerne l'émission et la susceptibilité aux phénomènes suivants, dans les avions militaires ; elles sont :
- incomplètes pour les pointes de tension subtransitoires (spikes)
- trop théoriques pour les transitoires de génération (surges)
- périmées pour les modulations et ondulations.

Le seul document existant qui fait une bonne approche de la compatibilité est le "DO.160" diffusé par le R.T.C.A. (Radio Technical Commission for Aeronautics) en collaboration avec l'EUROCAE (European Organisation for Civil Aviation Electronics). Ce document fixe ou recommande les procédures d'essais et donne les limites de fonctionnement normal et anormal des réseaux.

Pour les avions militaires, l'optimisation que nous recherchons nécessite plus de précisions, même quand certaines méthodes relèvent plus des "règles de l'art" que d'essais normalisés (ex : imprécision des impédances pour les phénomènes cités plus haut).

Zones de fonctionnement.

Tous les éléments des systèmes électriques resteront pendant la presque totalité de leur vie au voisinage des conditions de fonctionnement "Nominal", c'est-à-dire dans un domaine où les variations des paramètres sont connues quantitativement en amplitude et en périodicité, parce que systématiques pour l'accomplissement des vols ou des missions.

De part et d'autre de ce domaine, il reste encore des zones où le matériel doit conserver toutes ses performances mais où il se trouvera rarement, à la suite de défaut mineur ou d'utilisation marginale. Ces conditions peuvent durer pendant tout un vol mais ne se repéteront pas souvent. Ce sont des conditions éventuelles dans le domaine "Normal" ; on peut chiffrer leurs limites mais non pas leur durée. Cependant, elles affecteront peu le potentiel de vie des composants, ce dont on tiendra compte dans leur choix et les calculs de fiabilité. Exemples : tenue à des températures extrêmes, apparition et élimination d'un défaut sur un autre équipement, détérioration momentanée du filtrage, etc...

La troisième zone de fonctionnement, dite "Anormale", se situe au-delà de la précédente mais ne doit jamais dépasser des limites garanties par les protections générales. La sélectivité des protections doit être telle qu'elles n'agissent ni prématurément ni trop tard. Il en résulte que, suivant la gravité du défaut, le fonctionnement Anormal puisse se maintenir exceptionnellement jusqu'à intervention, c'est-à-dire au pire la fin du vol.

- On peut en conclure que les 3 niveaux de limites de fonctionnement, données par les normes citées plus haut, ne sont pas liés aux pourcentages de la puissance utilisée ou coupée mais à ce qui est dit ci-dessus. En outre, afin de mieux cerner les conditions de travail des équipements électroniques pour mieux les définir, il conviendrait de généraliser cette notion des 3 niveaux aux régimes stabilisés ou même sub-transitoires, au lieu de ne considérer que le normal et l'anormal.

Régimes.

- En régime stabilisé, l'alimentation électrique est définie par les paramètres Tension, Fréquence et Forme d'Onde. L'expérience a montré qu'il était plus facile de parer aux affaiblissements de la tension qu'à ses augmentations. Compte tenu de l'extrême rareté des tensions limites anormales, les normes les plus récentes citées plus haut (U.K., I.S.O., et DO.160) en ont remonté légèrement les valeurs ; ceci facili : la sélectivité des protections. Les valeurs correspondantes, données dans le tableau n° 2, sont celles des spécifications des avions français et donnent satisfaction.

La fréquence et la forme d'onde dépendront des réseaux (privilégiés ou non). Toutes les normes citées sont imprécises au sujet du domaine des fréquences en régime stabilisé. Pour les générateurs primaires à fréquence fixe, on peut exiger de ne pas sortir de la plage 390/410 Hz, se décomposant en :
4 Hz pour la modulation, 1 Hz pour la précision du réglage et 5 Hz pour le glissement ou dérive. Pour des réseaux différents, d'autres plages peuvent être définies contractuellement.

Les caractéristiques de modulation en alternatif et d'ondulation en continu sont imposées par les semi-conducteurs (de régulation, redressement ou conversion) qui peuvent ou non interférer entre eux. Elles n'ont plus pour origine les régulations non électroniques qui ont servi de bases aux limitations toujours imposées par les normes. Les amplitudes et spectres de fréquence de ces phénomènes sont à revoir en fonction de ce qu'ils sont réellement et des besoins des équipements. La bande la moins précise se trouve entre 100 et 1000 Hz. Le domaine de fréquence de ces perturbations rejoint donc largement celui des interférences électromagnétiques, transmises par conduction.

- Les coupures d'alimentation ou manques de tension peuvent être décomposées en :

-cas anormaux, moins de 7 secondes (normes) que l'on sait réduire de moitié, si nécessaire,
-cas normaux, moins de 200 millisecondes que l'on peut réduire à 50 ms sur des réseaux de faible puissance,
-cas spéciaux privilégiés (exemples systèmes calculateurs intégrés) que l'on peut garantir sans coupure à partir du réseau continu ou à moins de 50 ms à partir de l'alternatif, mais pour des puissances réduites.

- Les régimes transitoires.

Ce sont des variations entre 2 régimes stabilisés d'un paramètre ; leur durée peut s'étendre de quelques millisecondes à quelques secondes. C'est pourquoi on les appelle transitoires de longue durée ("Surges") ou de régulation ; ils proviennent du comportement de la génération et de ses réactions vis à vis des besoins du réseau. La puissance qu'ils mettent en jeu vient de la réserve ou de l'insuffisance d'énergie des générateurs.

- Les limites des transitoires de tension sont définies par les normes déjà citées suivant les différents niveaux de fonctionnement. Les courbes données n'y sont pas les enveloppes des transitoires, mais celles des créneaux d'énergie équivalente ("Step functions") qui risquent de surcharger les équipements utilisateurs.

Les valeurs des tensions maximales imposées sont réalistes, compte tenu de ce que les réseaux continus peuvent être alimentés par des génératrices ou des alimentations de terrain et que leur batterie peut être coupée. Par contre, si l'on est certain que cette alimentation se fera toujours soit avec les batteries de l'avion couplées, soit par les transfo-redresseurs de l'avion, les maxima exigés peuvent être abaissés chacun de 15 volts (soit maxi en anormal à 65 volts) pour le continu.

- Les limites que donnent les normes aux transitoires de fréquence sont des valeurs extrêmes rarement rencontrées sur les avions d'armes, en particulier la durée des valeurs anormales.

D'autre part, pour des durées inférieures à 2,5 ms (1 période du 400 Hz) ce n'est plus une variation de fréquence mais une déformation de la forme d'onde, due à la réaction d'induit dans une machine tournante ou à l'amorçage des circuits dans un système statique.

- Les phénomènes "Sub-transitoires".

Ces impulsions ou pointes de tension ("Spikes") de polarité quelconque se superposent localement aux tensions de chaque réseau. Elles sont dues à des phénomènes très brefs (haute fréquence) répétitifs ou non, venant des équipements ou de leur câblage, par conduction ou par induction. En raison des impédances variées de leurs sources, des lignes et des récepteurs pour les fréquences correspondantes, il n'existe pas encore de règles d'intervention applicables à tous les cas rencontrés.

Nous donnons ci-après un classement expérimental qui permet de fixer certaines limites et d'indiquer les protections recommandées. On distinguera :

- Les subtransitoires classiques (ou normaux) qui font l'objet de limitations dans les normes nationales, pour les réseaux continus seulement. Ils sont constitués d'une impulsion pseudosinusoidale plus ou moins rapidement amortie (fréquence autour de 100 kHz) qui met en jeu beaucoup moins d'énergie que les transitoires décrits plus haut (de l'ordre de 500 fois moins). Ils sont dangereux en raison de la valeur de la surtension atteinte (claquage) et de leur faible temps de montée (rayonnement). En alternatif, ils peuvent atteindre 1000 volts, ont un temps de montée de 2,5 à 5 micro-secondes et sont résorbés en moins de 10 micro-secondes. En continu, ils ne dépassent pas 600 volts mais la durée de l'impulsion est 2 à 5 fois plus longue. Nous avons reproduit à gauche de la planche jointe (en trait continu) les enveloppes de ces subtransitoires. La partie droite de cette figure donne les limites des transitoires de génération, en régime anormal.

- Les perturbations anormales, répétitives ou peu amorties, dont l'onde initiale est semblable aux précédentes mais pouvant atteindre respectivement 1200 V et 800 V ou se répéter par suite de résonnances ou d'un amortissement lent. La coupure brusque d'un circuit selfique en est un exemple courant. La fréquence de ces perturbations peut aller de 100 à 500 kHz. Leur enveloppe peut atteindre les limites en trait discontinu du centre de la planche citée.

- Les signaux à front très raide, même s'ils n'atteignent pas les amplitudes indiquées plus haut peuvent parasiter les équipements utilisant des composants modernes (genre bascule électronique) ou des informations digitales. Des mesures ont montré que le rayonnement d'une ligne, siège d'une variation de tension de 500 à 700 V en 100 nanosecondes, nous mettait dans ce cas.

- D'autres signaux non maitrisés peuvent se propager dans les réseaux de bord en atteignant localement des valeurs prohibitives. Ils peuvent avoir différentes causes (faisceau Radar, impulsions Laser, décharges électrostatiques, émissions électromagnétiques, etc...) L'expérience montre que sur les avions actuels, ils ne dépassent qu'exceptionnellement les enveloppes "normales" décrites plus haut. Mais un avion qui serait débarrassé, à leurs sources, de tous les "Subtransitoires" d'origine réseau pourrait être encore le siège de tels phénomènes.

Applications pratiques :

Tous les subtransitoires anormaux (autres que ceux dits classiques ci-dessus) qui peuvent apparaître localement dans un réseau, plus ou moins amortis par les impédances de ligne, devront faire l'objet de recherches et de parades, soit par amélioration de câblage, soit par mise en place de dispositifs de protection écréteurs.

Remarque : la figure de la planche jointe, tracée en double échelle logarithmique (hyperboles d'amortissement transformées en lignes droites) permet de rassembler les limites demandées et d'apprécier les écarts d'énergie à écouler suivant leur durée. L'excès de puissance mise en jeu dans chaque cas dépend de la surintensité, donc de l'impédance de source et de charge pour la fréquence du phénomène transitoire.

3.2. Equipements et "Avionique".

Compatibilité avec les réseaux.

Tous les équipements électriques et électroniques doivent être conformes aux normes nationales des Caractéristiques des réseaux électriques de bord. Jusqu'à présent, la compatibilité à ces exigences n'était pas codifiée par des procédures d'essais. Les exceptions citées au début de ce chapitre (Normes existantes) s'avèrent intéressantes mais ne sont que des "exigences minimales".

Des documents constituant des recommandations communes sont indispensables pour la validité des essais de compatibilité "Equipements - Réseaux". Leur mise au point est complexe et devra présenter assez de souplesse pour être applicable à des niveaux d'exigences propres aux avions militaires et mentionné dans les spécifications contractuelles (ex. Radar).

Les conditions d'environnement et des essais de qualification doivent être précisés dans les spécifications particuliéres ou contractuelles des fournitures. En particulier, pour les conditions d'antiparasitage magnétique et électromagnétique, par conduction et rayonnement, émis et reçus, la norme la plus complète à ce sujet est la MIL.STD 461 A (Notice 3/USAF)
complètée par la MIL.STD 462.

Alimentation.

Les normes classent les équipements en catégories A, B, C,... Z suivant leur utilisation. Celles-ci n'étant pas connues à priori, ne se sont pas imposées. L'expérience montre que pour les chutes de tension la quasi-totalité est dans la catégorie "B" (c'est-à-dire 4 volts autorisés en alternatif et 2 volts en continu). Pour éviter tout malentendu, il convient de ne mentionner ces classes que pour les cas d'espèce, dans les spécifications particulières (ex. classe ou sévérité d'essais d'environnement).

L'utilisation de plus en plus abondante d'électronique impose des précautions contre leurs interférences diverses, il faudra :

- Eviter au maximum les courants de circulation par le neutre de l'alternatif. Pour cela, les utilisateurs de courant alternatif auront une consommation équilibrée sur les 3 phases, câblées sans neutre.
- Seuls les utilisateurs de quelques dizaines de V.A. pourront être monophasés à condition que la phase soit indifférente.
- Le neutre, comme le Zéro volt du continu seront câblés, isolés l'un de l'autre et du boitier.
- L'écran des circuits blindés, interne ou externe, sera raccordé à une borne de connecteur pour continuité des blindages.

Fonctionnement.

- En accord avec les zones de fonctionnement décrites plus haut, (en 3.1.) les équipements conservent leurs performances dans toute la zone de fonctionnement "Normal", qu'il soit systématique ou éventuel. Ce dernier cas plus dur mais rare n'affectera pas souvent le potentiel de vie des composants.

- Entre la limite maximale précédente et la limite extrême du fonctionnement "Anormal", les prescriptions des normes s'appliquent (performances dégradées ou déclenchement). Les normes ne précisent ni si, en cas de déclenchement spontané d'un équipement en conditions anormales, le réarmement est automatique ou non, ni si après un fonctionnement anormal maintenu une vérification de l'appareil s'impose. Les spécifications du système devront le préciser.

- Le fonctionnement "Secours" (Emergency) concerne essentiellement le fonctionnement sur batterie en courant continu, qu'on limitera à 18 volts aux bornes de l'équipement (au dessous de cette valeur, ce n'est qu'un régime transitoire ou cas d'espèce exceptionnel).

- Le tableau n° 2 (déjà cité) rassemble les limites de tension des types de fonctionnement envisagés. Les valeurs qui y sont soulignées se retrouvent dans toutes les normes, les autres valeurs plus récentes sont conformes aux documents B.S.I. / I.S.O. et DO.160.

Toutes autres limites devraient faire l'objet de documents particuliers contractuels, résultant de l'accord des parties. Les normes ainsi allégées ne seraient plus sujettes à interprétations.

Comportement vis à vis des perturbations.

- D'une façon générale, tous les équipements électroniques et électriques doivent pouvoir fonctionner en présence des perturbations dont le niveau ne dépasse pas les conditions dites "Normales" (ou classiques), qu'il s'agisse d'ondulations, modulations, parasitage électro-magnétique, transitoires ou sub-transitoires. Inversement, ils ne doivent en aucun cas générer dans le réseau des phénomènes de mêmes niveaux (mesurés à leurs bornes). Pour la susceptibilité aux Sub-transitoires atteignant 600 ou 1000 V, il s'agit de niveaux, réglés à circuit ouvert avec une impédance de source déterminée (20 à 50 ohms suivant procédure) avant d'être appliqués à l'équipement. Il existe d'autres procédures au moins aussi exigentes (voir DO.160.).

- Pour obtenir les résultats ci-dessus, excepté le cas des subtransitoires examinés plus loin, et pour protéger les circuits dans la zone des transitoires "Anormaux", on utilisera des filtrages convenables et un dimensionnement suffisant des composants. Se reporter aux limites en trait continu figurant sur le côté droit de la planche jointe (on pourra abaisser à 65 V les maxima en courant continu).

- Les équipements devront se protéger eux-mêmes contre les Sub-transitoires, dits classiques plus haut, de même qu'eux-mêmes ou leurs composants ne devront émettre de pointes de tension de mêmes niveaux (limites en trait continu figurant à gauche de la planche). Ceci pourra être obtenu au moyen de dispositifs écréteurs:

> Les montages générateurs d'impulsions devront les étouffer au moyen de "Varistances" et non avec de simples diodes qui modifient le temps de réponse et la tenue des relais.

La simple protection contre les pointes plus brèves venant du réseau sera mieux assurée par des combinaisons de "Transzorbs" (réponse plus rapide). Les autres perturbations anormales ou à front très raide en dehors des limites ci-dessus doivent faire l'objet de parade dans les réseaux de l'avion.

4. METHODES DE DISTRIBUTION.

Quel que soit le soin apporté à la réalisation de chaque réseau électrique d'un aéronef, des impératifs de sécurité et de masse ou d'encombrement conduisent à des limitations d'utilisation.

Quand la puissance électrique installée en générateurs primaires dépasse une vingtaine de kVA, il convient de séparer les réseaux en 2 systèmes indépendants (au moins). Chacun d'eux pourra comporter :

- des circuits de sécurité, récupérables en cas de perte des sources primaires,
- un réseau principal conservé tant que les sources primaires de l'avion permettent de l'alimenter,
- des circuits auxiliaires ou "délestables" qui, en cas de besoin, permettent de conserver la sécurité de l'avion au détriment de sa mission.

Un certain nombre de circuits spécialisés, éventuellement symétriques, peuvent être greffés sur les réseaux ci-dessus. Le choix des fonctions affectées à chaque système dépend de l'emploi de l'avion, ex : Vol - Armement pour un avion militaire ou Droite - Gauche pour un avion civil. Ces choix permettent, le cas échéant, une ségrégation des implantations d'équipements, des câblages et de leur parcours, etc...

Suivant le nombre disponible de générateurs, on sera amené à les coupler en parallèle, 2 à 2 s'ils peuvent aller par paires, ou à transférer tout ou partie des réseaux de l'un sur l'autre en cas de besoin. Ces couplages ou transferts peuvent être confiés à de simples logiques, confirmées ou non par l'équipage, ou à des ensembles de surveillance et de protection automatique. Il n'y a pas de règles absolues, l'évolution des techniques donnant à chaque série d'avions une solution propre.

[illegible], ce qui se retrouve similaire dans tous les schémas électriques d'avion est le principe de [illegible] des protections : les protections principales, chargées d'éliminer un générateur ou un réseau [illegible], ne doivent pas être influencées par les défauts dont l'élimination est confiée aux protections [illegible]

Ceci est obtenu par des différences de calibres, de niveaux de tension ou des temporisations dont l'échelonnement impose les zones de fonctionnement décrites plus haut (3.1.). Ces dispositions, confirmées par des essais et des études de pannes, permettent de chiffrer les risques de passage des zones fonctionnelles aux zones anormales. Les résultats de ces études comparés à celle de la fiabilité des équipements doivent permettre leur optimisation.

5. CONCLUSIONS.

Afin de contribuer à une meilleure utilisation de l'avionique et des équipements électriques, nous suggérons que pour améliorer l'approche des différents problèmes, il faudrait :

- Alerter les différents constructeurs sur l'importance primordiale de la définition des conditions de fonctionnement. Ne jamais déroger aux normes sans entente préalable sur un compromis. Examiner de part et d'autre, et ensemble s'il y a lieu, les impératifs qui semblent contradictoires.

- Etudier avec réalisme les conditions limites d'utilisation (ex : fréquences rares ou exceptionnelles) et les chiffrer (importance et risques de se produire) de façon à choisir des composants ou équipements dont le potentiel de vie soit optimal.

- Réviser un certain nombre d'habitudes, en mettant l'accent sur quelques points : Réduction des consommations de courant continu (réservé aux réseaux de secours) et des consommateurs de courant alternatif à fréquence fixe (400 Hz) régulée, réservé aux systémes de calcul ; enfin meilleure connaissance et diminution maximale des perturbations apportées aux autres consommateurs.

TABLEAU 1

EVOLUTION PRESUMEE DUE A LA FREQUENCE VARIABLE

Supposed evolvement due to variable frequency

EQUIPEMENTS	PUISSANCE A FREQUENCE VARIABLE Concerned power	VARIATION DES GRANDEURS EN % Change rate of values				
		POIDS Weight	VOLUME Volume	PRIX Price	FIABILITE Reliability	PERTES Losses
ALTERNATEUR Generator 390 à 1050 Hz (6 pôles)	80 %	- 35	- 35	- 65	+ 20	- 60 à 65
TRANSFO REDRESSEUR T.R. (4 KW)	5200 VA	+ 5 à + 10			- 5	+ 5
8 POMPES CARBURANT (Pumps)	15200 VA	+ 20	+ 10 à 15	+ 30	- 10	+ 10
ENGINS	50 à 80 %	Faible (a few)	0	+	0	- 32 à 48
SYSTEMES SPECIAUX	33 à 75 %	+ 5	+ 15	+	0	- 21 à 45
RADAR (Doppler)	65 %	+ Filtres	+ 15	Les plus modifiés Most changes		- 42 à 39
BROUILLEURS	2800 VA	PEU DE CHANGEMENTS (a little)				

Générations électriques et réseaux de bord
dans les avions modernes

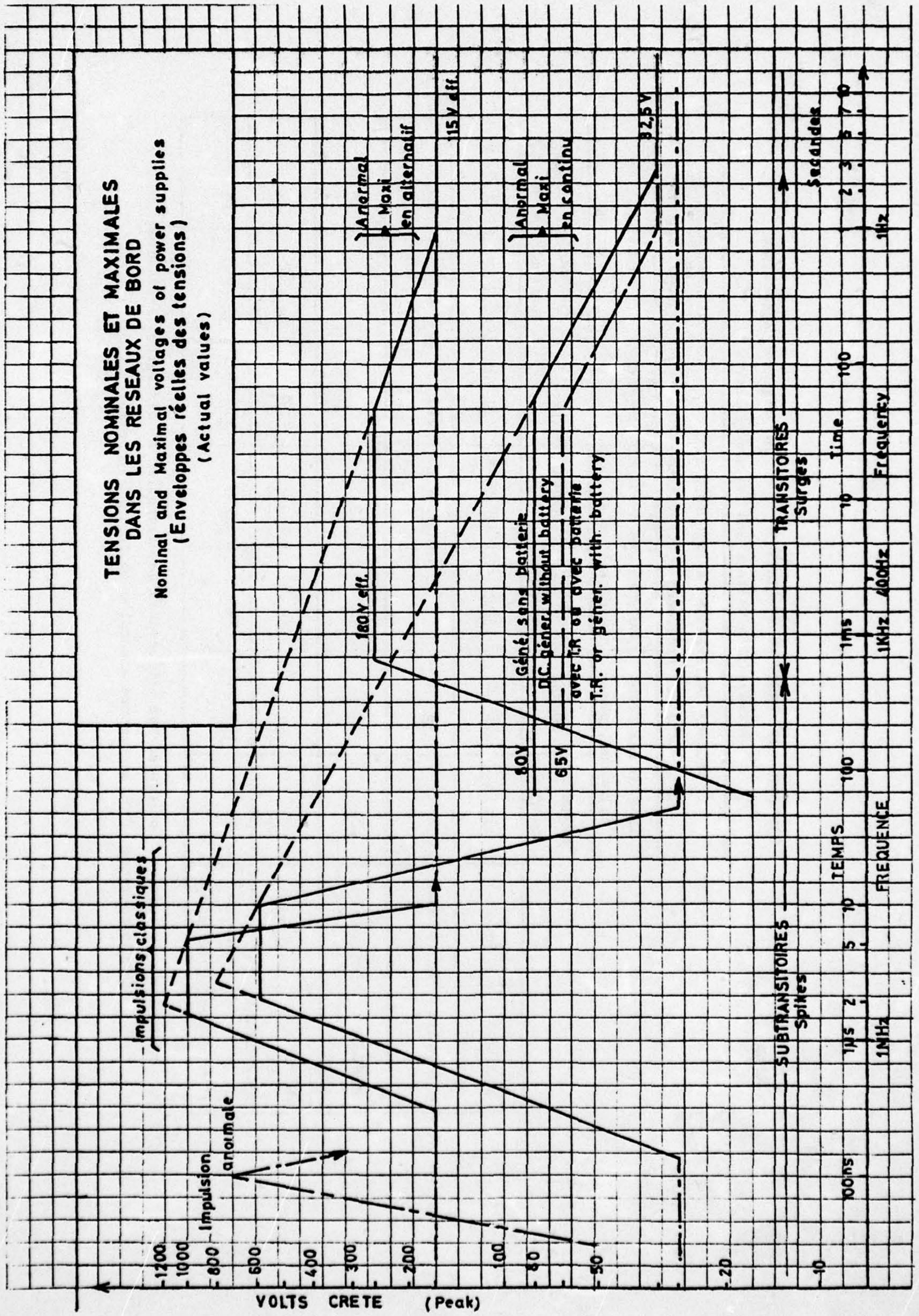

TABLEAU 2 TENSIONS ET FREQUENCES LIMITES en régimes stabilisés

Steady state voltage and frequency limits

		RESEAUX "ALTERNATIF" A FREQUENCE FIXE Constant frequency A.C. systems					RESEAUX "CONTINU" D.C. systems	
		TENSION PAR PHASE Individual phase		MOYENNE DES 3 PHASES Average of the 3 ph.		FREQUENCE	TENSIONS Voltage	
		ANORMAL abnormal	NORMAL	ANORMAL abnormal	NORMAL	(Hertz)	ANORMAL abnormal	NORMAL
MAXIMUM général (Everywhere)	Rare	132 V	122 V	130,5 V	120,5 V	420 Hz	32,2 V	30,3 V
	Fonctionnel		118		116,5	410		29,5
MINIMUM	barres-bus (bus bar)	106	112	108	113,5			26
MINIMUM aux équipements (at terminals)	Normal (1)	102	108	104	109,5	390 (380)	20,5	24
	Secours (Emergency)	97	100	99	102	360 (Maxi 440)	15	18

(1) **NORMAL aux équipements correspond aux équipements de catégorie "B" des normes existantes**

" **is for category "B" equipments according to present standards.**

DISCUSSION

N F J Allum:

On behalf of ISO I would like to tell you that although ISO 1540.2 has been balloted, highly significant comments were received and it will be re-balloted as ISO 1540.3. This will then align with the latest RTCA agreement.

A Beau:

Thank you. I believe there is little difference between the voltages given in the two documents ISO 1540.3 and RTCA (DO 160).

AIRCRAFT POWER SUPPLIES & COOLING PROBLEMS -

A VIEWPOINT FROM THE POWER CONDITIONER DESIGNER

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Kent

SUMMARY

The paper details the main trade-offs in a modern avionic power conditioner designed to interface between electronic units and Aircraft Power Supplies. The ability to overcome the major limitations of these supplies as specified by BS3G100 etc. is demonstrated and the advantages to be gained by improving them are discussed.

It is proposed that a systems approach, rather than the consideration of power supplies and/or power conditioner alone, will produce a better solution to the thermal problems associated with avionic equipment.

1. INTRODUCTION

The advance in the development of aircraft power supply systems has lagged behind the advance of avionic systems which during the past ten years, have multiplied and grown radically in complexity. The power supply specifications meanwhile have changed very little. This is not altogether surprising since the avionic power load is a relatively small percentage of the overall generator load (approx. 10% on a transport aircraft). Power Conditioner technology has therefore advanced to meet the demands of a high technology avionics system operating from a power source which is designed to supply high powers to heating, lighting and motor loads.

It is important that the problems and the trade-offs implicit in meeting aircraft specifications are understood in order to determine where improvements are needed to optimise avionic systems.

This paper has four objectives:-

1. To outline the major limitations of aircraft power supplies as they are at present defined by specifications such as BS3G100, MIL-STD-704-A, SP-P-90001, etc.
2. To show the capability of modern technology to overcome these limitations.
3. To outline the potential benefits of improving the primary power supply quality to both power conditioner and avionic system.
4. To determine possible power saving alternatives with present and future systems.

2. POWER SUPPLY LIMITATIONS

2.1 Surges and Transients.

The major limitations of aircraft supply specifications are the surge variations as defined in BS3G100, MIL-STD-704-A etc. and 'drop-out' transient. (See Fig. 1). The transient duration is too long to consider energy storage as a means of overcoming these for any but the smallest equipment: hence, the power conditioner must either operate over the full range, or the unit will cease to function.

2.2 Harmonic Distortion on a.c. Supplies

For all but the larger units the volume of components necessary to meet the harmonic distortion specification (Fig. 2) is excessive. Hence, only larger units (greater than 1kW) attempt to meet this specification. It is unlikely that an a.c. filter could reduce the harmonic distortion sufficiently and produce an input impedance which would meet the power factor limit. This is because the capacitance allowed across the a.c. lines is severely limited by the power factor specification. The calculated value of capacitance, assuming a resistive load, is 1 μF/kW Line-Line, which is equivalent to 3 μF/kW phase-neutral.

3. CAPABILITY OF POWER CONDITIONERS

3.1 a.c. Supply Transients.

Power conditioners have undoubtedly concentrated on overcoming the input surges whilst attempting to meet increasingly stringent efficiency, weight and volume specifications. A typical computer specification is shown in Fig. 3. In this case the input specification is Panavia SP-P-90001 and the operational minimum is 72 V rms per phase less a 2 V line drop.

The specification was met by an off-line PWM bridge inverter in a volume of 1640 cm^3. Figure 4 shows the block schematic and Fig. 5 shows a photograph of the final mechanical construction of the unit. This circuit configuration was adopted for the following reasons:

1. Step down regulators and boost regulators place a common L-C filter between the output(s) and the supply, hence crosstalk between outputs is dependent on the frequency response of these regulators. The PWM inverter does not suffer from this disadvantage.

2. The four-transistor inverter is superior in its use of transistor voltage and current parameters.

3. Since power control and isolation are performed in one circuit the bridge inverter offers good prospects for minaturization.

The equivalent circuit of the PWM bridge inverter simplifies to a step-down switching regulator where the output voltage is the average value of the transformer secondary voltage, i.e.

$$V_o = nV_i . t_{on} . \frac{2}{tp} \qquad 1.$$

where t_{on} is determined by the input voltage. Obviously the smaller the ratio of $t_{on}:t_p$ the larger the 'off-time' and the greater the energy storage requirement. The volume of an inductor is roughly proportional to $\frac{1}{2}LI^2$ with V_o/nV_i given by equation 2.

$$E_L = (1 - V_o/nV_i) . \frac{V_o I_o t_p}{2k} \qquad 2.$$

The value of capacitor is dependent upon the ripple current passing through it (kI_o) and the output ripple voltage allowed (mV_o) - equation 3.

$$E_c = \frac{k}{m} \cdot \frac{P_o t_p}{16} \qquad 3.$$

The total energy storage capability is the sum of equation 2 and 3.

$$E_T = (1 - V_o/nV_i) \cdot \frac{P_o t_p}{2k} + \frac{k}{m} \cdot \frac{P_o t_p}{2} \qquad 4.$$

The relationship between energy storage and volume is extremely complex and depends on the type of capacitor used the voltage derating the voltage rating, and the type of inductor material, whether powdered iron or gapped steel. Inductors, however, tend to be larger and heavier (and more expensive) than capacitors for the same energy storage, hence it is the inductor that should be reduced, and could be if the voltage range were to be reduced.

The other factors determining filter volume and weight are ripple current, output power and frequency. The ripple current flowing in the inductor (kI_o) determines the minimum output current that the filter can deliver before the filter regulation increases significantly, as shown in Fig. 7. Volume can also be seen to be proportional to output power and inversly proportional to frequency.

The inverter frequency is within the control of the designer and is chosen to be high enough to reduce the weight and volume of filters and transformers and minimise audio noise, but low enough to maintain reasonable switching losses and control range, the lower end of which is limited by transistor storage time. A reasonable compromise for a wide range converter is 20 kHz, the approximate size of a 200W transformer being a 35mm cube.

In the conditioner under discussion the parameters were calculated as follows:-

Allowing a 2 μsec off-time at maximum 'on' time, i.e. lowest input voltage, will produce an average voltage at the transformer primary of 155V and, allowing for voltage drops and ripple, 140V is a sound design average transformer primary voltage.

The minimum 'on' time is given by:-

$$\frac{V_o t_p}{V_\phi \sqrt{6}} \qquad 5.$$

which in this case equals 8 μsec, and the nominal 'on' time will be 13 μsec.

The theoretical ratio of stored energy to output energy of the 5V supply is therefore approximately 12:1. The major problem with practical filters in avionic conditioners is the lack of suitable capacitors, that is capacitors with an equivalent resistance low enough to maintain reasonable levels of output voltage ripple and with 125°C operating temperature in a small volume. In order to meet the specification of Fig. 3 capacitors were required with a resistance of less than 40 mΩ (100mV ripple voltage and 2.5A ripple current) and a combination of polycarbonate and wet tantalum was chosen.

The wide input range also affects the choice of power transistors. These are required to handle the maximum input of 440 volts ($180.\sqrt{6}$) at a current of nearly twice the nominal average input current. In comparison with an inverter operating from a transient-free supply, the transistors of this converter require a Vce Imax product 2.5 times greater. This reduces transistor availability, but not, at this power level, size or weight.

Efficiency is also reduced as the ratio V_i nominal : V_i minimum is increased. Resistive losses, up to the secondary rectifiers, increase as the square of V_o/nV_i and switching losses linearly with V_o/nV_i.

The size of the EMC capacitor is also proportional to V_o/nV_i but design of this filter is complicated by the negative resistance load presented to it by the regulator.

The unit has since been modified to provide 5V at 16A, 30V at 2.5A and ± 15V at 1A, a total of 180 Watts within the same outline. Fig. 8 demonstrates the improved maintainability attained with a similar design, which delivers 150W in 2460 cm³ and uses plug-in printed circuit boards and modules.

3.2 D.C. Supply Transients

The 28V d.c. supply surges are more severe than those encountered on the a.c. supply. Fig. 1 shows the maximum as being 80V and the minimum as 16 volts, or in some cases 7V. At these low voltages the Boost Regulator circuit has a number of advantages and, combined with a high frequency converter to provide isolation and multiple outputs, has proved most successful. Fig. 9 shows a prototype 80 Watt unit built on two metal-cored printed circuit boards with an overall volume of 980 cm^3, and an efficiency of 80%.

The boost regulator circuit has been analysed a number of times [1,2]; in general, these analyses are simplified in the following way:-

Feed-forward control of loop gain is normally omitted, as are considerations of non-linearity of the inductor. Since the former is almost essential to maintain stability, with minimum component volume, and the latter is useful for reducing inductor size, these are important techniques. A degree of experience is therefore necessary, when designing this type of circuit, if optimum results are to be achieved.

With an upper surge limit of 80V, the circuit must boost to at least this level to maintain satisfactory operation, unless the following converter is shut down. At this level the operational current levels within the converter are significantly lower than would be experienced if a step down pre-regulator were used. A reduction of 6:1 would result if a 16V operational minimum input voltage were required. Once again analysis of the converter circuit is not difficult to find, and components to operate at this level are readily available.

The volume and weight of the boost filter are again complex functions of $\frac{Vo}{Vi}$. The energy stored in the inductor increases in accordance with equation 6 which is shown plotted in Fig. 10.

$$E_L = \frac{P_i t_p}{2k} \cdot (1 - V_o/V_i) \qquad 6.$$

The CV product of the capacitor decreases slowly with decreasing output voltage and it is this as opposed to the energy storage $\frac{1}{2}CV^2$ to which the volume is approximately related. The CV product is given by equation 7 and plotted in Fig. 11.

$$C_o V_o = \frac{P_o t_p}{m} (V_o - V_i)/V_o^2 \qquad 7.$$

In the two designs outlined above, the zero voltage transient was disregarded and the unit allowed to shut-down. This is generally allowed in the system specification: the exception is systems essential to flight safety, such as flight control, engine control, stores management etc.

These systems must maintain full-time operation and, with the use of the 28V d.c. supply, support is available from the aircraft battery during a transient. A second method of surviving transients to meet MTBF requirements, is consolidation of supplies and computer lanes. With this technique a double failure (at least) is necessary before the zero-voltage transient affects the unit.

Further improvements will be forthcoming with the introduction of semi-conductor Bus Bar Switching. This is expected to reduce the zero transient by a factor of between 5 and 10.

3.3 Harmonic Content of Conditioners Operating from a.c. Supplies

The volume of the inductors and capacitors necessary to reduce the harmonic distortion, by filtering, to limits within specification are frequently prohibitive. However, other methods of overcoming this problem on a 3-phase supply exist. First, a regulating technique is under development which significantly reduces line current distortion and produces regulated d.c. outputs. This method is complex but could be attractive for loads in excess of 1kW, and utilises the basic power constancy of balanced 3-phase resistive loads. The second method does not regulate, but buffers the avionic conditioners from the 3-phase supply by the use of 24-phase rectification. Whilst it may not be practical to use one buffer unit per equipment it would be feasable to buffer a number of equipments with a more powerful unit, thus gaining size and weight advantages. Such a system followed by a high-frequency regulator and inverter is comparable in size to a 400Hz isolating transformer. Fig. 12 shows the line current and voltage waveforms of a prototype unit. As can be seen the harmonic content of the transformer line current is less than 8% which is reduced further by the RFI suppression filters. (Compare with Fig. 2c where the distortion is approximately 30%).

Other Regulating Methods

There are a number of other techniques, in addition to those detailed above, capable of overcoming some or all of the limitations of the aircraft supplies. The use of ferroresconant [3] phenomena is one such technique, the attractiveness of which would be increased if the input voltage transient range were to be reduced. The high frequencies, and hence lower weight and volume, possible with resonant turn-off circuits would be fully exploited if the supply were to exhibit a reduced transient range.

4. IMPROVEMENTS AND THEIR IMPACT

Some of the advantages of an improved supply specification have been mentioned above. It is necessary to determine the extent and type of improvements required and the advantages thus gained. Ideally the equipment designer requires the power system to provide the voltages necessary to power his circuits at the terminals of his equipment. This, however, leads to a proliferation of wires within the aircraft and has been shown to be impractical in a number of cases.[4] A reasonable alternative would be a high voltage (100v) dc busbar regulated to approximately 5%.[5,6] Below this level linear regulation would become a viable alternative to switching types within the user equipment. However, the line drop between the pre-conditioner and the avionic load would be an essential parameter and 7-10% pre-regulation at the equipment terminals seems more likely than 3-5%.

If, however, the supply variations were reduced to $\pm$ 7% at the equipment terminals then conditioners would be considerably lighter and smaller, and the efficiency would be improved, but to a lesser extent. Fig. 13 shows the probable optimum efficiency of a 5V switching regulator incorporating isolation. The difference between the efficiency of this unit (81%) and the efficiency of the previous regulator (75%) is 6%, with 10% as a possible maximum, that is a saving of between 12 and 20 Watts in 200.

However, it may not be necessary to redesign the power supply to achieve these advantages. The input voltage is below 105V (Fig. 1) for less than 150 msec and although aircraft stability systems, engine controller and similar systems must remain operational during this period, a number of units not essential to aircraft safety could safely shut-down. With orderly drop-out, information can be stored in low-power stores so as to avoid corrupted data when the system is reactivated.

It is relatively easy with a switching regulator to supply power for, say, 100 - 200 μ sec during a drop-out in order to organise the storage of essential information. Fig. 14 shows the components used to supply 75 Watts at 5V for the orderly shut-down of a computer core store.

As can be seen it is possible to store 80mJ of energy and, with a suitable switching regulator, easily use three quarters of this. A similar system was proposed for a 28V d.c. system. The total power involved was only 20 Watts but it was required to operate during a 50 msec zero voltage transient. In this case full advantage could be taken of electrolytic capacitors and a 350 μ F 100V capacitor was required to store the 1Joule of energy necessary.

Another very important method of saving not only power but weight and complexity is to avoid over specifying the outputs of the power conditioners. It is all too easy for engineers to specify the conditioner to the same Periodic And Random Deviation (PARD) as the bench supply that was used for the breadboard. Switching regulators have a far worse transient response than a Linear Regulator (because of the L - C filters involved). Hence the combination of a tight voltage tolerance and large transients will frequently restrict the designer to the use of a linear regulator with the attendant penalty of reduced efficiency.

Various estimates of the savings possible with improved power sources have been made.[6] Our own estimates are shown in Table 1.

TABLE 1

	PSU Contribution to Avionic Unit	Avionic Unit Reduction Due to Improved Power Source
Weight	25 - 35%	10 - 20%
Volume	20 - 30%	10 - 15%
Cost	10 - 30%	5 - 15%
Power	25%	5 - 8%

This assumes that a wide-range conditioner is replaced by a unit designed to operate with 5% negative transient.

Finally, the obvious place to initiate power saving is at the load. 20 Watts saved downstream of the conditioner is 25 Watts saved at the power source. Thus, the use of CMOS Logic, Class D amplifiers

power alternatives I would suggest the system specification should be questioned.

5. CONCLUSION

It can be seen that, whilst the large transient variations of aircraft supplies and the harmonic distortion limit do not make the design of power conditioners easier, these limitations can be overcome with present day technology, but with attendant weight, volume, cost and timescale penalties.

By far the greatest advantages would accrue if switching regulators could be replaced by linear regulators or by non-regulating convertors. This implies a d.c. voltage source of around 100V with a PARD of not greater than $\pm3\%$. A carefully designed linear regulator with an isolating high frequency converter inside the loop could achieve 80% efficiency and provide cost, timescale, weight and volume advantages significantly greater than those outlined above. This could be an attractive long term aim. In the short term the unit designer and system designer need a greater awareness of the problems of power conditioning. In engineering terms more feedback is required to obtain optimum results. The philosophy that, because a ½-ATR box is capable of handling 200 Watts with a cold wall cooling system, one should actually dissipate 200 Watts within this size is an admission of defeat. Power usage requires optimisation in the same manner as other system parameters if the advantages outlined above are to be gained.

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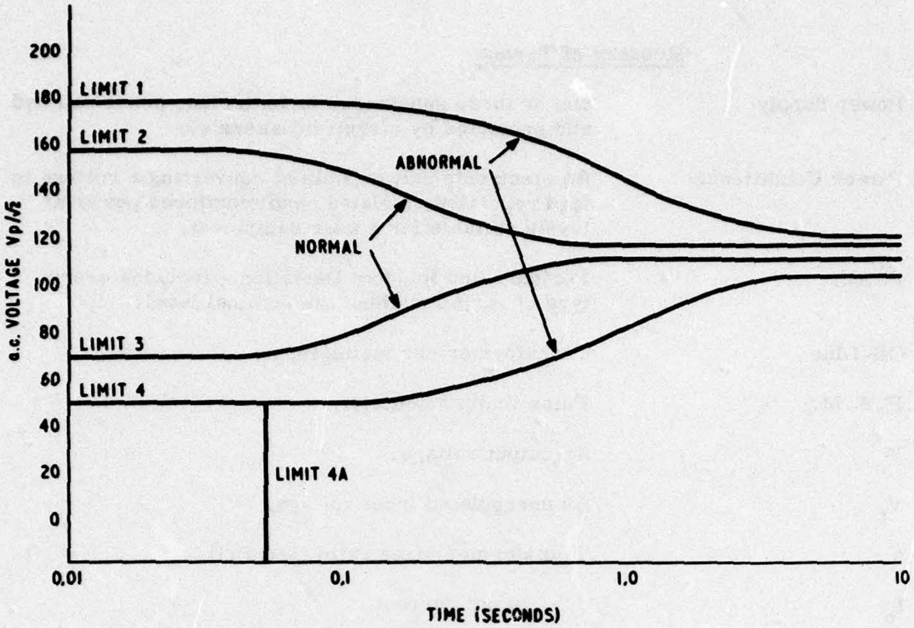
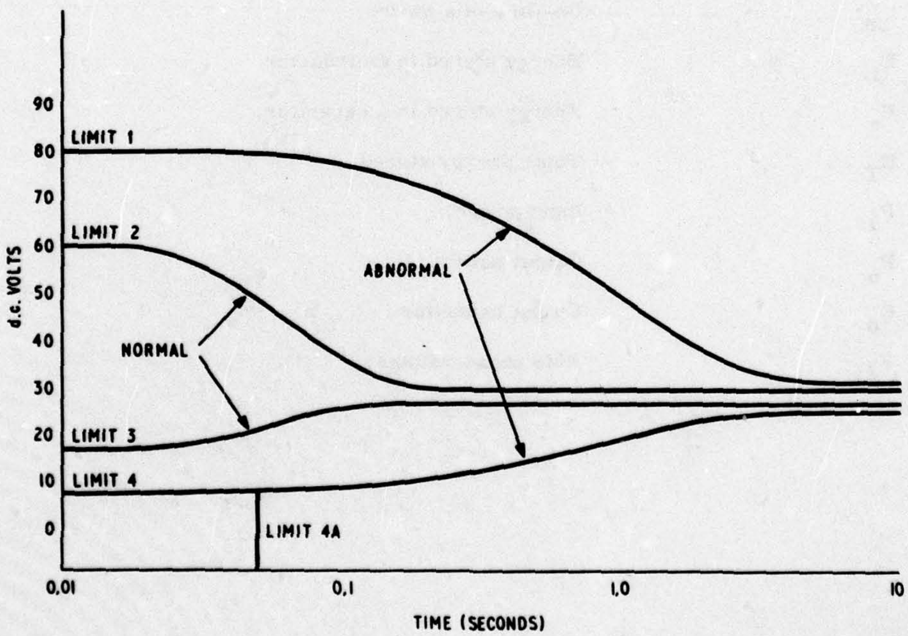
Glossary of Terms

Power Supply:	one or more generators or batteries, cross coupled and protected by circuit breakers etc.
Power Conditioner:	An electronic unit capable of converting a voltage to feed regulated, isolated, and monitored power at levels suitable for a user equipment.
PARD:	Periodic And Random Deviation - includes every type of variation about the nominal level.
Off-Line:	Transformerless mains input.
P.W.M.:	Pulse Width Modulator.
V_o	An output voltage.
V_i	An unregulated input voltage.
n	Transformer turns ratio (Sec:Pri).
I_o	d.c. output current.
k	p-p Ripple current/d.c. output current.
m	p-p Ripple voltage/d.c. output voltage.
f	Frequency
t_p	$1/f$
t_{on}	On-time of a switch.
E_L	Energy stored in an inductor.
E_c	Energy stored in a capacitor.
E_T	Total Energy stored.
P_i	Input power.
P_o	Output power.
C_o	Output capacitor.
V_ϕ	RMS phase voltage.

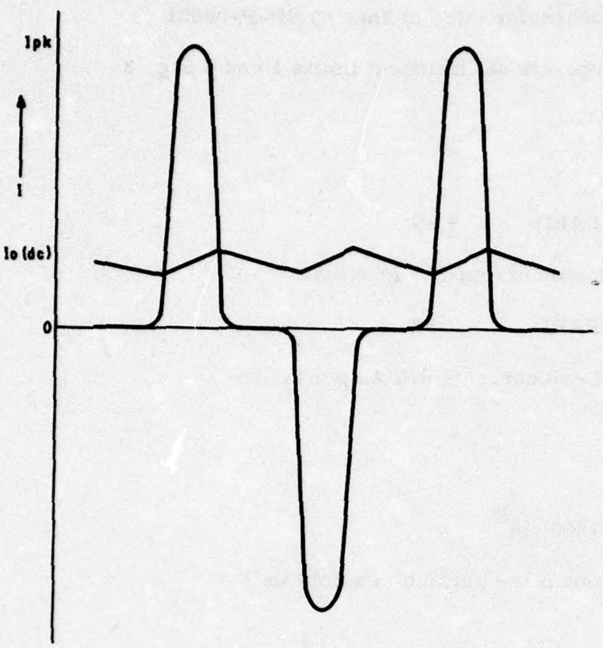


3 phase 400Hz system

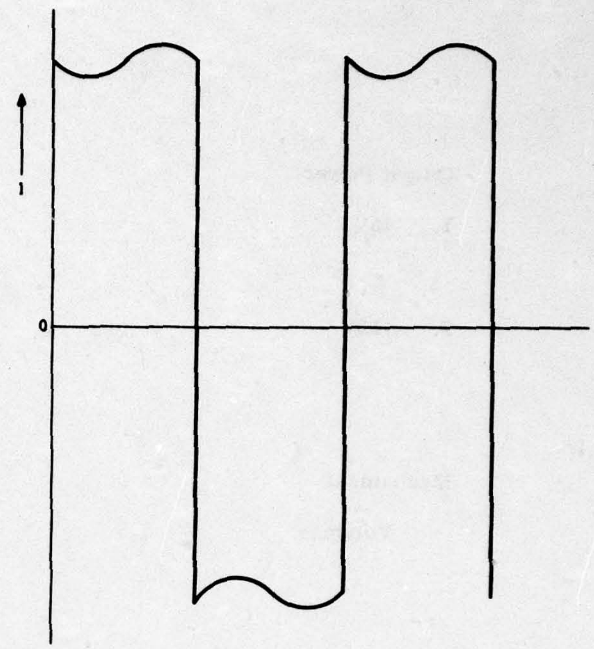


28V dc system

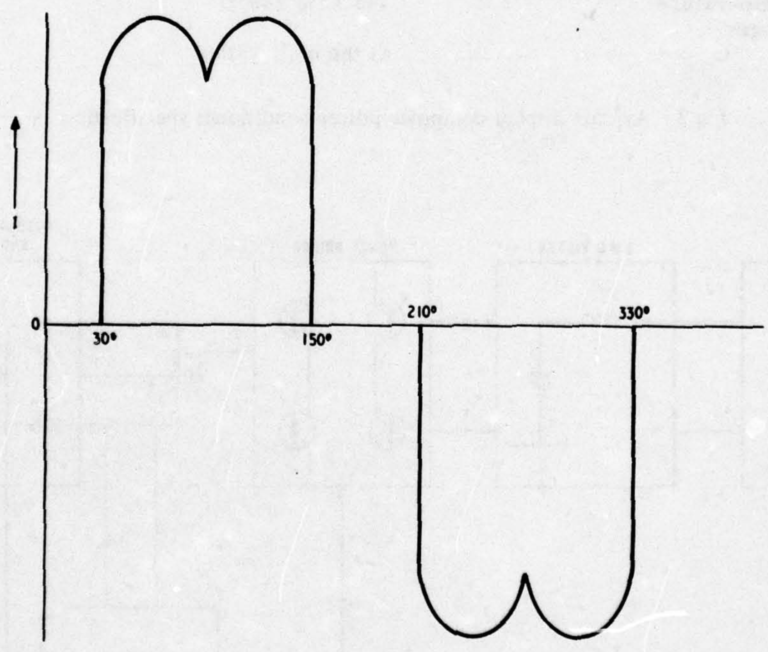
Fig.1 Equivalent step function loci of transient voltages as specified in SP-P-90001



(a) Single phase Capacitive input filter



(b) Single phase inductive input filter (50% distortion)



(c) 3 phase full wave - resistive load (30% distortion)

Fig.2 Line current waveforms of various filter circuits

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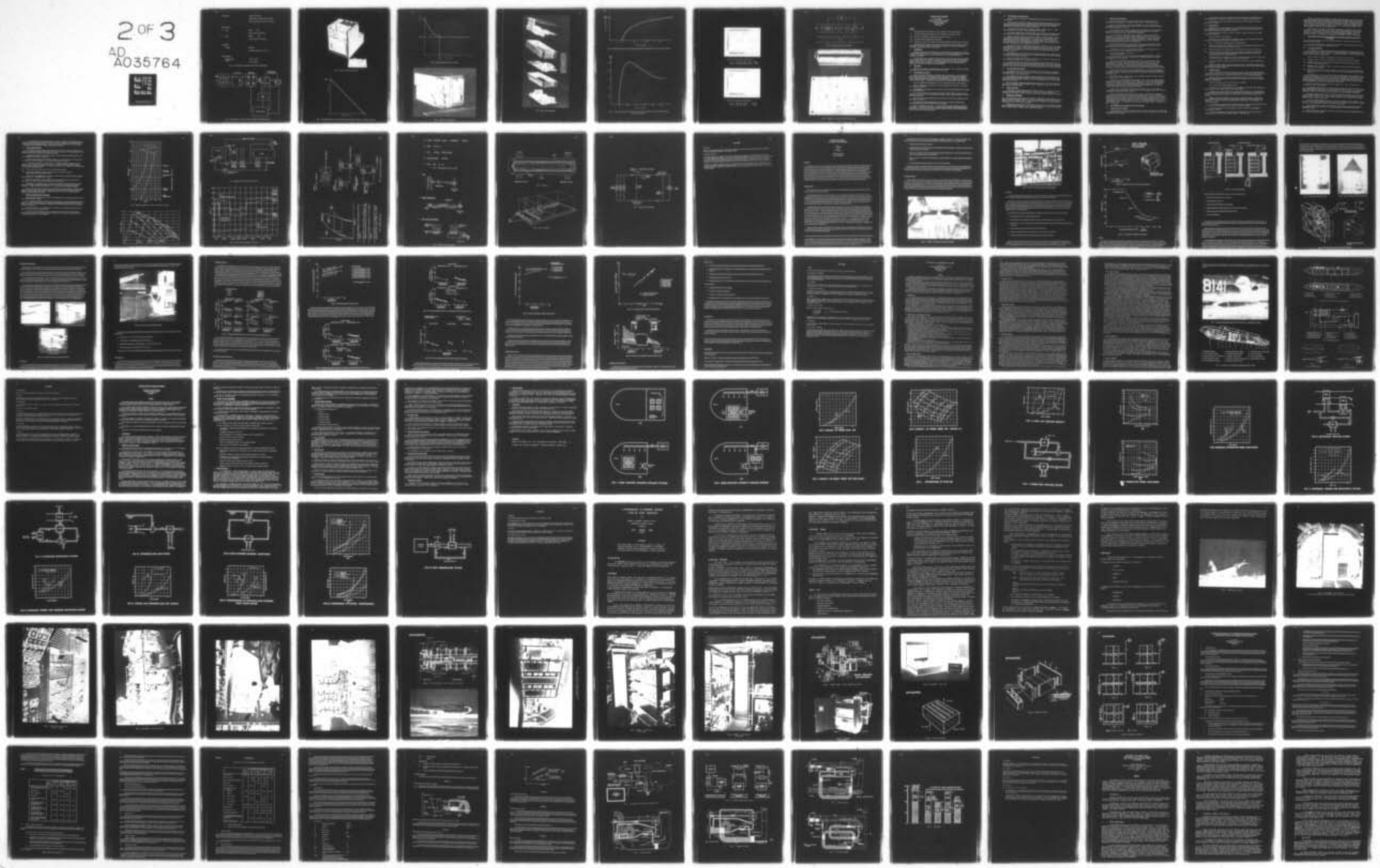

Input Power	**3-phase 115V/200V**
	characteristics defined by SP-P-90001
	operational transient limits 1 and 3 Fig. 3
Output Power	
1. +5V	**PARD $\pm$ 5%**
	Load current 25 Amps max
2. $\pm$12V	**PARD 5%**
	Load currents 0.5 Amp max
Mechanical	
Volume:	**1600 cm^3**
	***one plane surface* to a cold wall**
Environmental	
Temperature Range:	**-55°C to +85°C**
	at the cold wall

Fig.3 Avionic display computer power conditioner specification

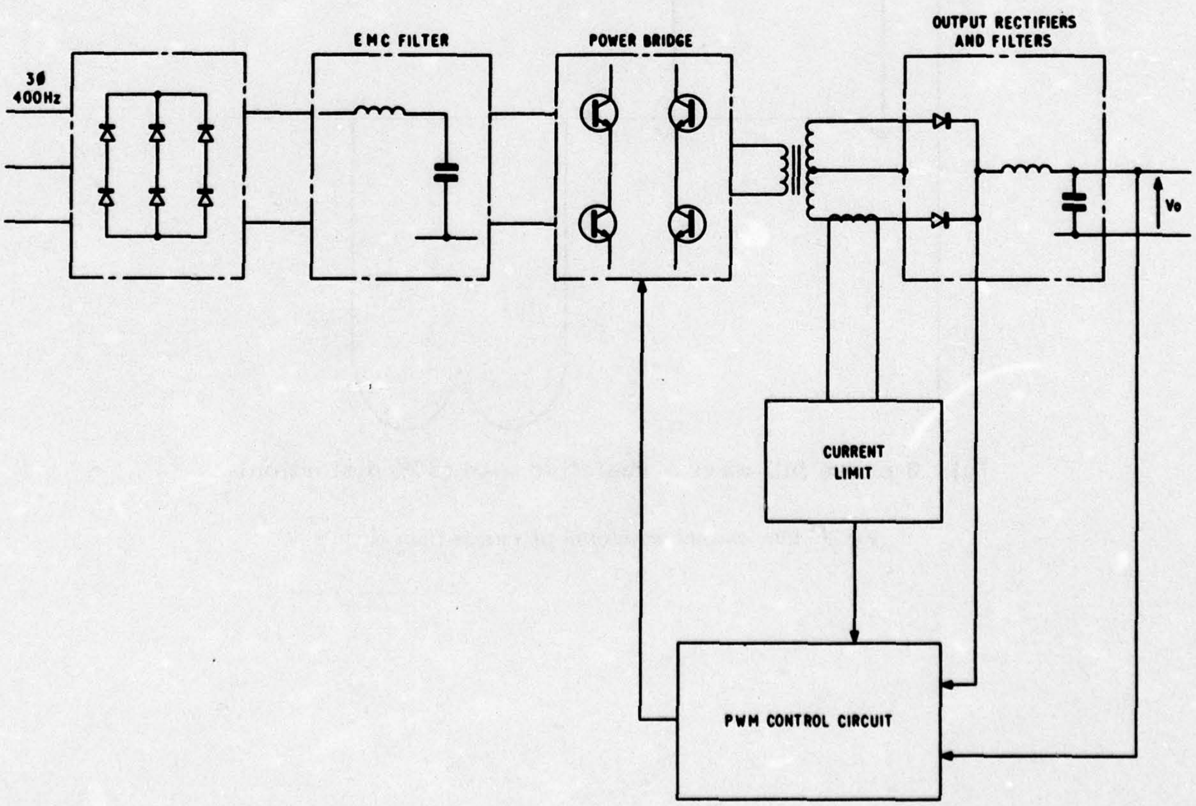

Fig.4 Block schematic of power conditioner utilizing a PMW bridge inverter

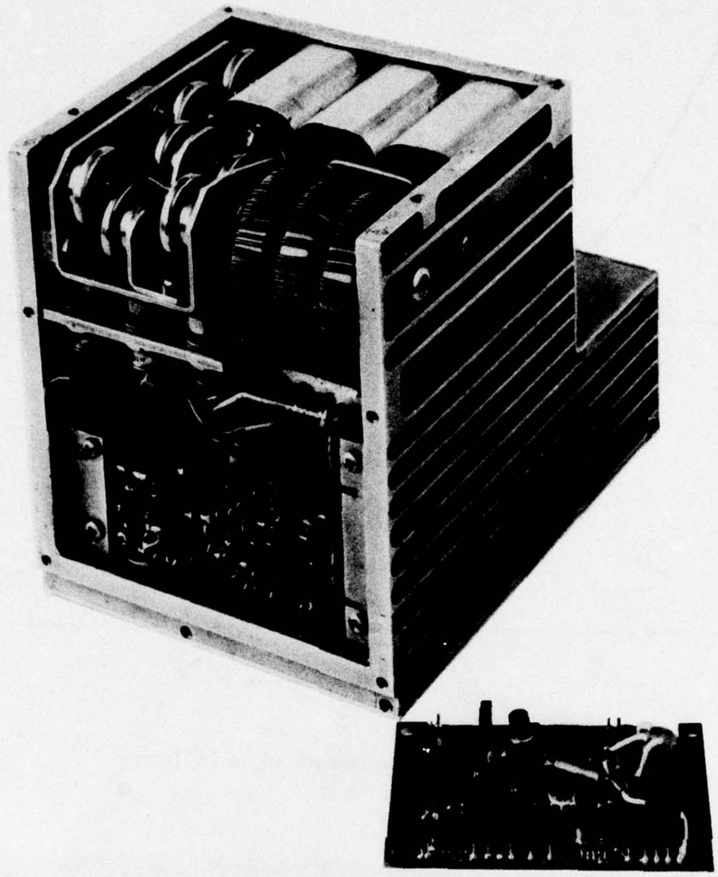

Fig.5 150W Avionic power conditioner

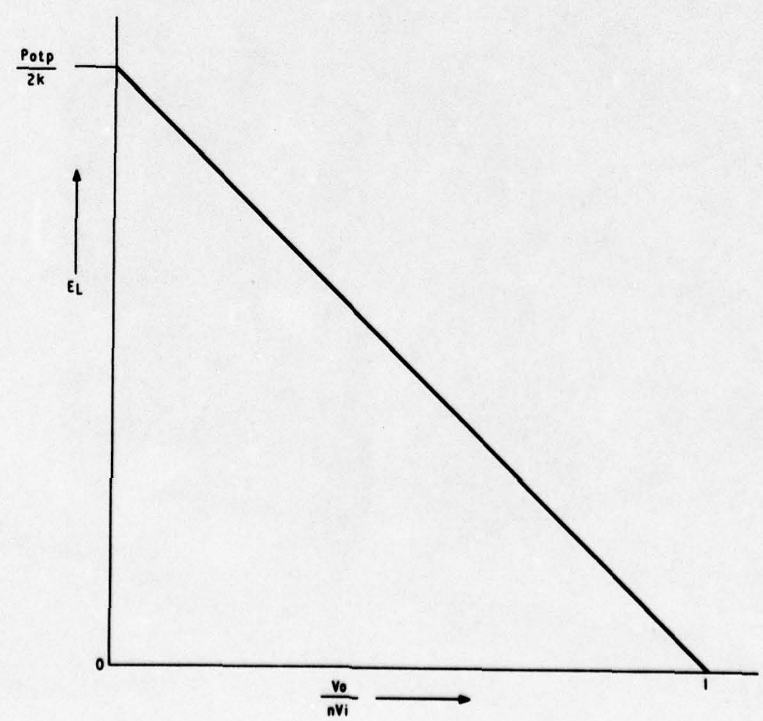

Fig.6 Relationship between the energy storage and input/output voltage ratio - stepdown converter

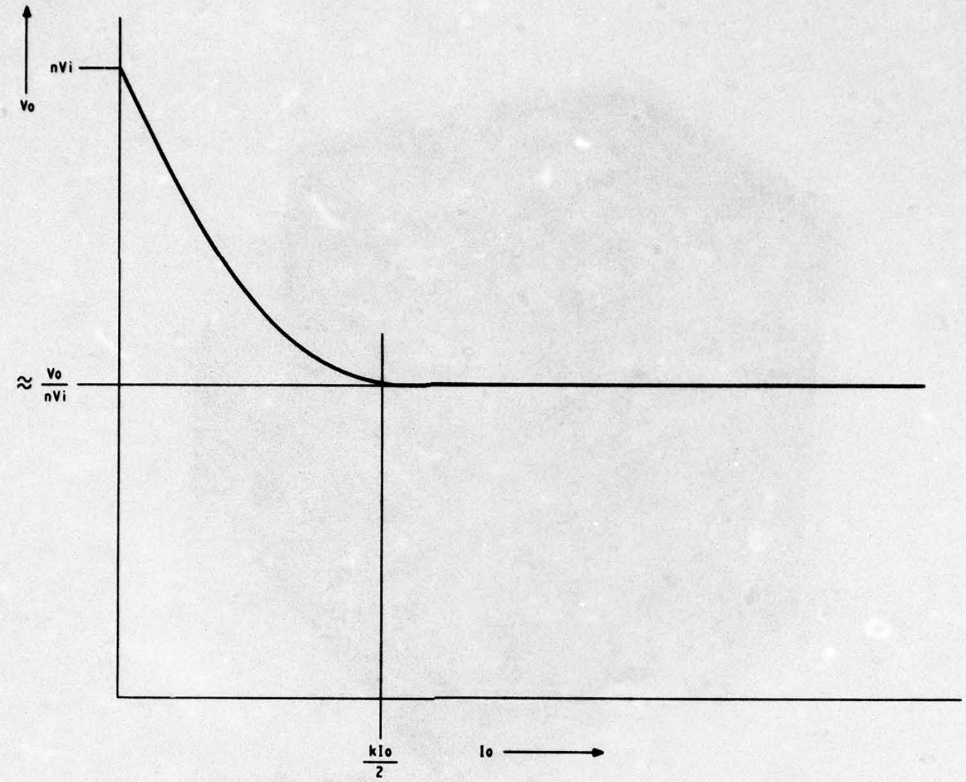

Fig.7 Regulator characteristic of an LC filter

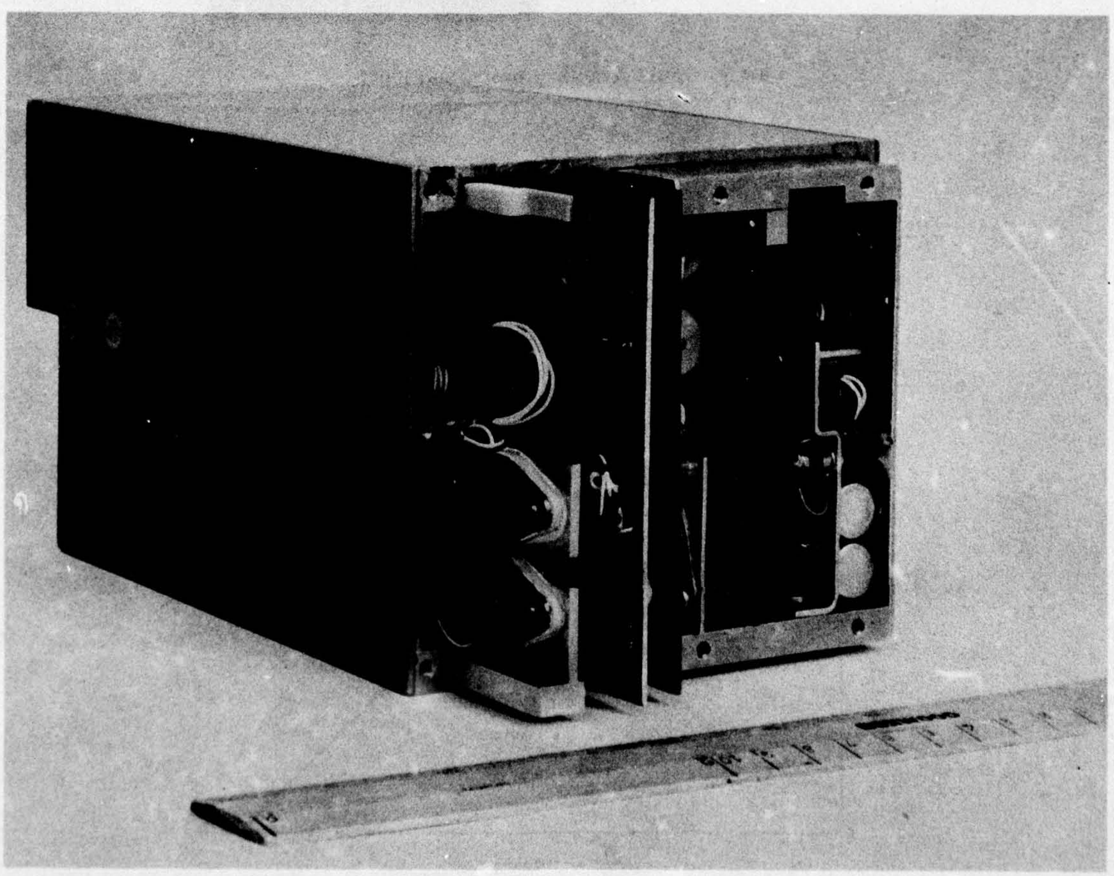

Fig.8 Modular power conditioner

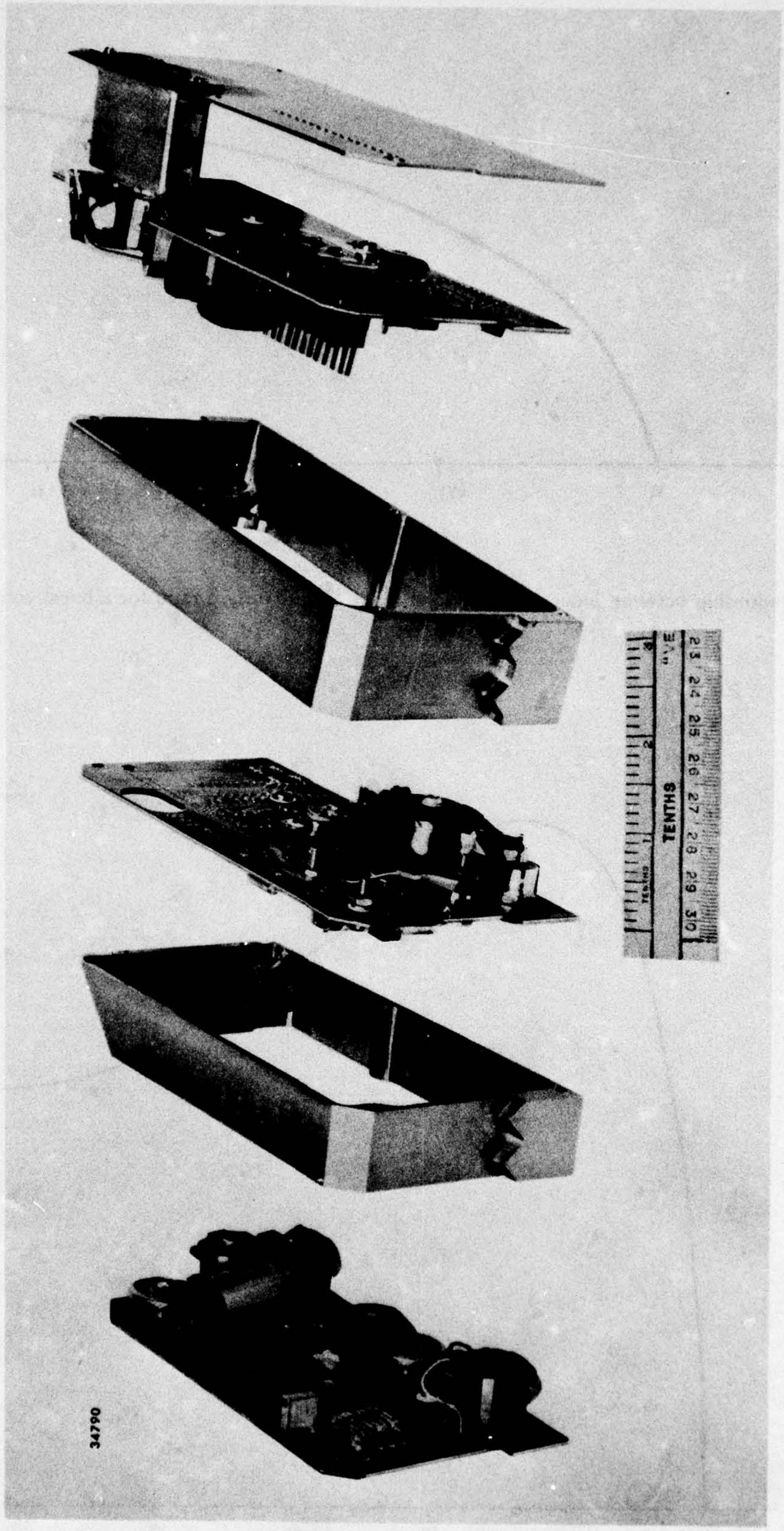

Fig.9 DC/DC power conditioner

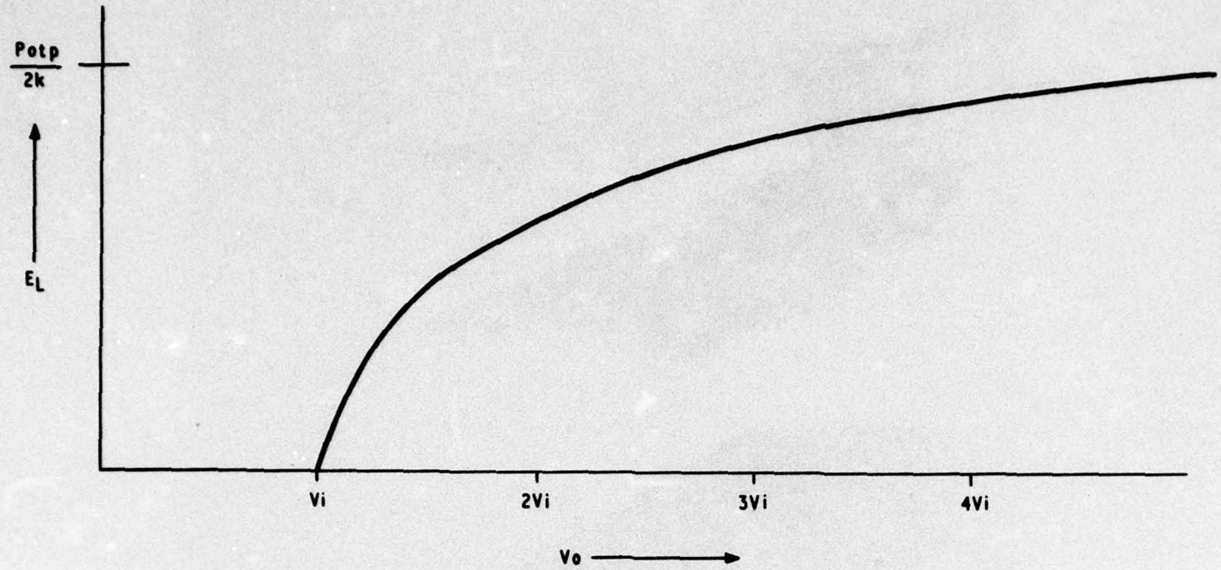

Fig.10 Relationship between inductor energy and input/output voltage ratio for a boost converter system

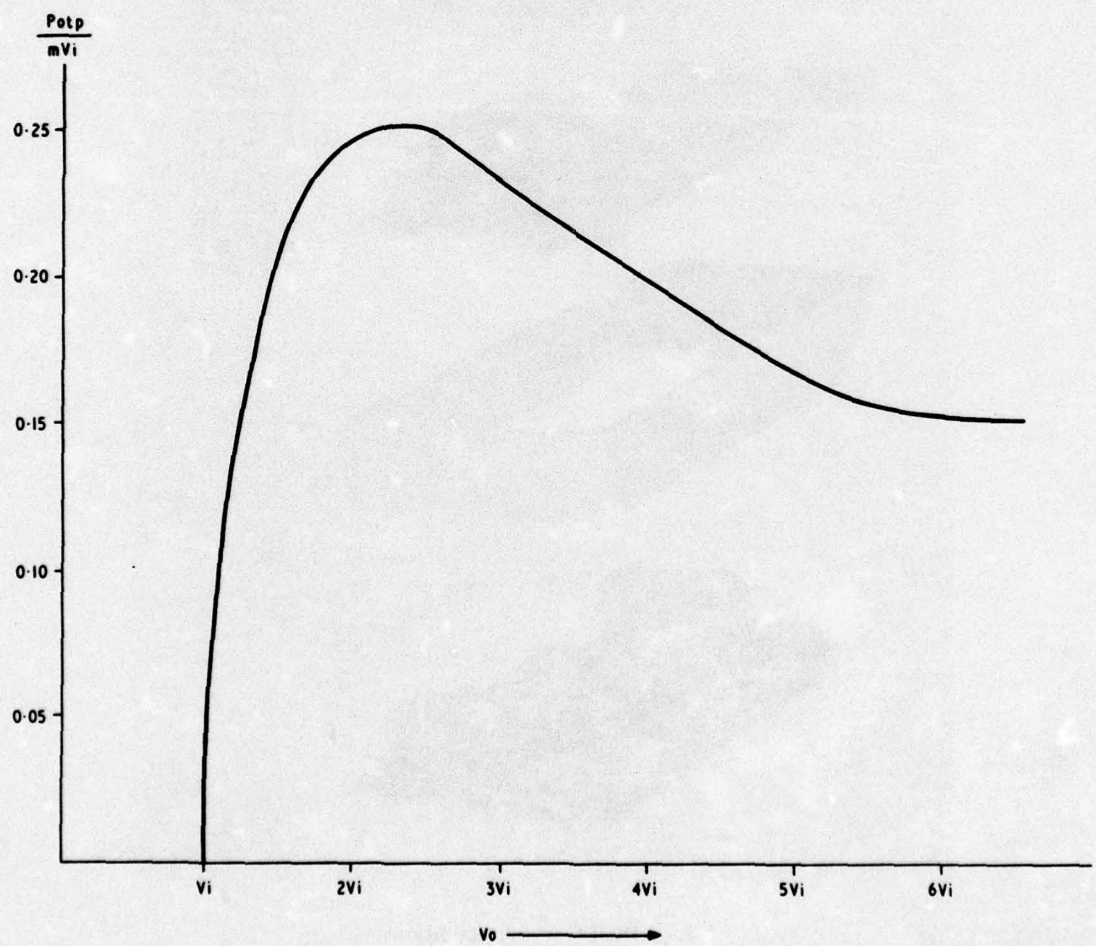

Fig.11 Relationship between the CV product of the output capacitor and the output voltage from a boost converter system

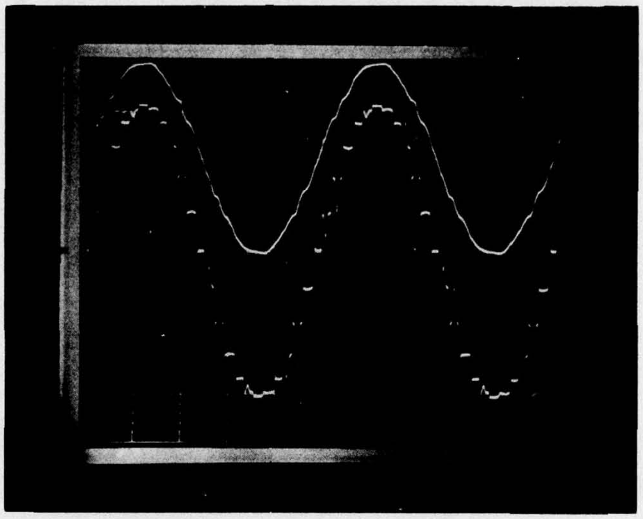

Fig.12a Waveforms prior to EMC filtering

Top:	Transformer phase voltage	80V/div
Bottom:	Transformer primary current	2A/div

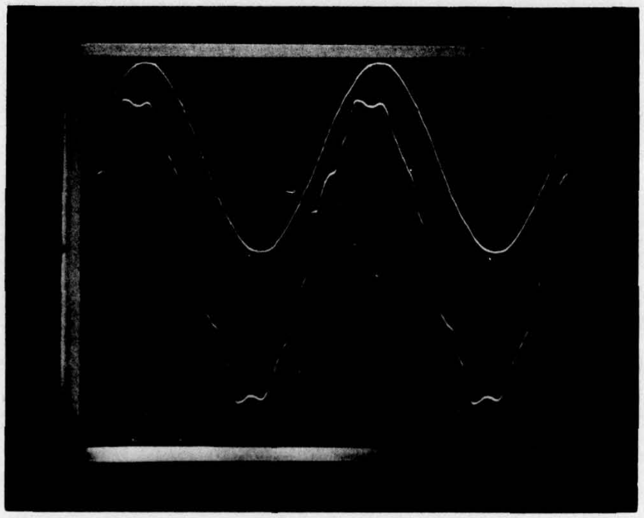

Fig.12b Waveforms after EMC filtering

Top:	Input phase voltage	80V/div
Bottom:	Input line current	2A/div

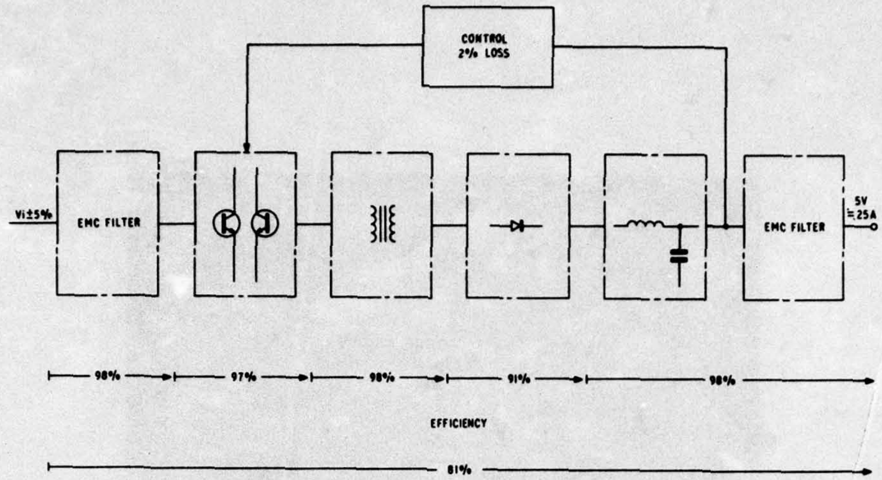

Fig.13 Optimised efficiency of a 5V SMR

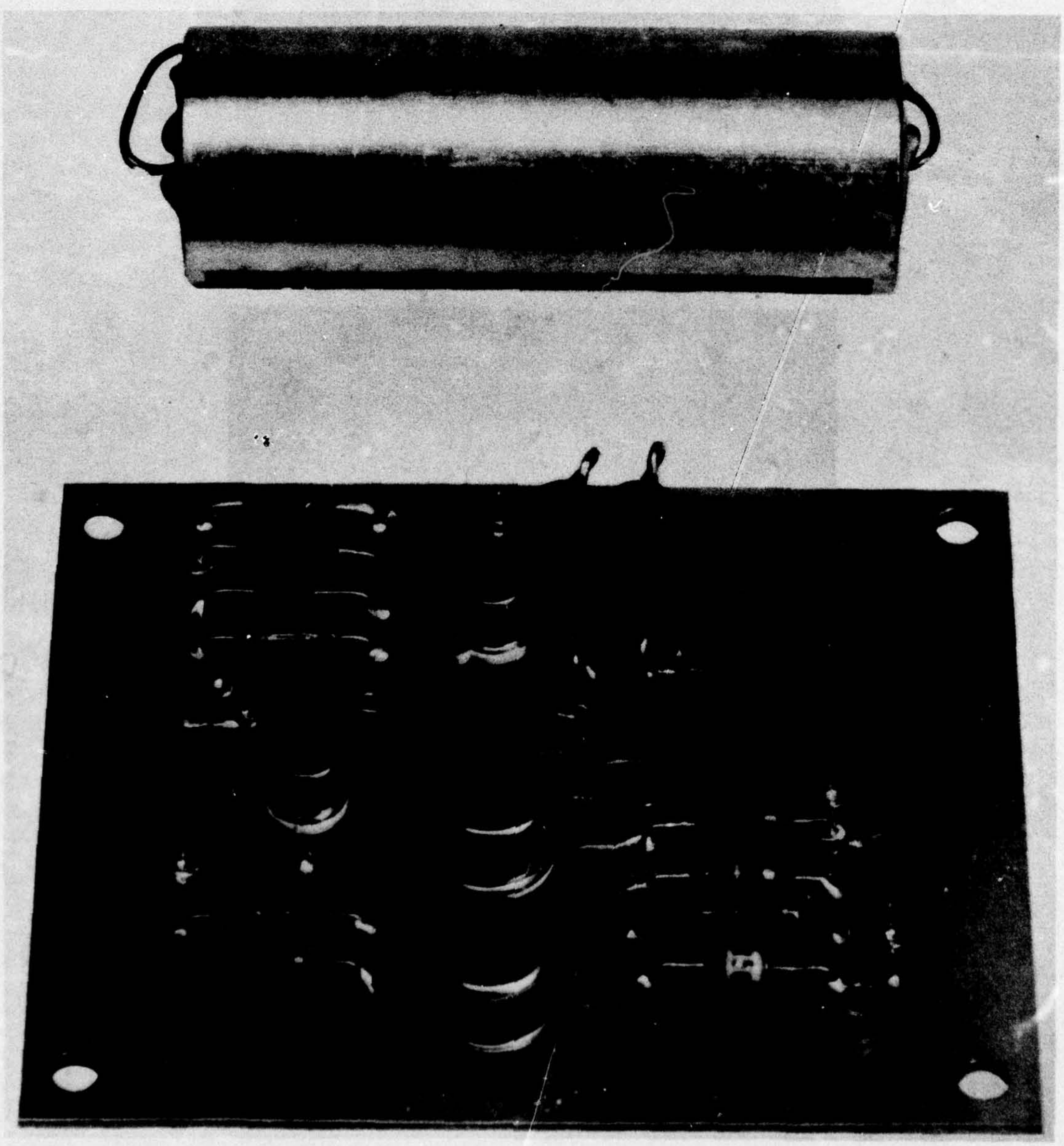

Fig.14 Components to provide a short term energy store

AIRCRAFT COOLING TECHNIQUES

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SUMMARY

This paper gives broad information on and discusses the following topics:-

(i) The basic aircraft heat sinks and their suitability for systems cooling;

(ii) The use of refrigeration techniques in military aircraft;

(iii) Cooling and heat transportation techniques;

(iv) Application to cooling of electrical and avionic equipment installations in aircraft.

Although the main content of this paper is purely of an informative nature, the main conclusion is drawn that plenty of scope exists for improving the cooling effectiveness in avionic and electrical systems installations.

1. INTRODUCTION

Because of the large heat rejections, and the inevitably high penalty on the aircraft of rejecting this waste heat, it is extremely important that the most economical use possible is made of the available heat sinks and therefore, the cooling techniques must be continually developed.

In this paper, we will examine the compatibility of the major heat sources in military aircraft with the available heat sinks, describe the techniques available for refrigeration and the methods which can be used for transporting the heat from its source to the cooling medium. The objects of the paper are to give information, but also to promote useful discussion on the topic of aircraft cooling techniques.

2. HEAT SINKS

There are two basic heat sinks available on the aircraft - ram air and fuel, although expendable heat sinks could also be carried for limited use.

2.1 Heat Rejection to Ram Air

Ram air cooling is the process of rejecting heat from the source to the external air flow around the aircraft. This generally involves taking air from the aircraft boundary layer or the region adjacent to the boundary layer and passing it through heat exchangers. By the addition of heat to this coolant air, its temperature is increased, but because of system pressure losses, its momentum is reduced and this invariably adds to the aircraft drag.

Let us now examine the suitability of ram air for direct use as a cooling medium.

Fig. 1 illustrates the variation of ram air temperature with Mach. No. at sea level and tropopause conditions.

Superimposed on these curves are the absolute maximum operating temperatures for major heat source components.

It can be seen that for many of the heat sources, direct cooling by ram air is not possible. In fact, ram air is a significant heat source itself (kinetic heating) in a high performance aeroplane.

The only really feasible possibilities of using ram air directly are engine oils and some types of alternator.

The other major problem associated with using ram air concerns the variation of available pressure (pressure differential) and air density.

Fig. 2 illustrates this point. At very low aircraft speeds insufficient pressure differential is available to move the air through the heat exchanger without employing some form of pump. At high altitude, the air density becomes very low, grossly reducing the air mass flow and hence its cooling capacity.

2.2 Heat Rejection to Aircraft Fuel

Fuel cooling is the process of transferring heat from the heat source into the aircraft fuel.

This is generally achieved by passing the fuel through heat exchangers as it flows from the fuel tanks to the engines, although one heat source, the fuel pumps themselves, reject their heat directly into the fuel by virtue of the "flooded motor" design of electrically driven pumps which are commonly used today.

A typical fuel cooling system (in simplified form) is shown in Fig. 3. This shows how the fuel can be used to cool some typical heat sources.

As a cooling medium, fuel is vastly superior to air, because of its higher density and specific heat, giving greater heat capacity and higher heat transfer coefficients.

However, problems do exist in fuel cooling systems. The most obvious of these is the variation in fuel flow with engine power, i.e. there is a large reduction in fuel flow when the engines are throttled back. In transient cases, this can largely be overcome by means of fuel recirculation back to the fuel tanks. This necessitates the use of special mixers as the recirculation fuel returns to the tanks, in order to avoid excessive stratification of fuel temperature in the tank.

Care has to be taken to prevent fuel boiling in the recirculation line, particularly when using AVTAG type fuels. It often becomes necessary to use an air cooler in the fuel recirculating line to prevent a rapid rise of the tank's fuel temperature when the tank fuel level is low. This can only be used, of course, at low speed flight when ram air temperature permits it.

The temperature of the fuel at the start of a mission depends very largely on the fuel storage methods used, and in particular, the ambient temperature and solar conditions to which the fuel bowser is subjected.

Typical fuel temperatures during an aircraft sortie for an aircraft using its fuel heat sink to the practical limit are shown in Fig. 4.

It can be seen that the temperature limits for fuel at the engine burner, and for cooling hydraulic oil are reached. The really significant points to notice are the gradual increase in fuel collector box temperature throughout the sortie and the rapid temperature rise which could take place during the final taxy, unless auxiliary air cooling is provided.

2.3 Use of Expendable Heat Sinks

An alternative to the fuel or air heat sinks is to carry a quantity of expendable heat sink material. This can take the form of a liquid (e.g. water) whose latent heat of evaporation gives the heat sink capacity.

In the majority of cases, the amount of heat sink required for the particular job would necessitate an intolerable quantity of heat sink material to be carried, and so this method is not normally employed. (0.64 kg. of water is required per kW. hr, assuming 100% evaporation efficiency).

The only practical cases where expendable heat sinks can be used are for very small heat loads (in particular those with special cooling requirements), or for heat loads of a short duration transient nature.

2.4 General Conclusion

At this point the most important general conclusion which can be reached is that for the more temperature sensitive heat sources, e.g. cabin and avionics, some form of refrigeration is necessary in order to provide a heat sink of adequate quality and quantity.

3. REFRIGERATION TECHNIQUES

3.1 For the purpose of this paper, by refrigeration techniques, we are referring to the methods which can be employed in obtaining a heat sink medium suitable for cooling an aircraft cabin or conventional avionic system. Special devices, for example those giving cryogenic temperatures, are not dealt with in this paper.

Refrigeration methods which can be employed in aircraft form a very extensive subject. However, this paper only attempts to take a rather shallow look at the subject, and discusses only the aspects which are thought to be of immediate relevance.

3.2 Vapour Cycle Refrigeration

Vapour cycle refrigeration is the way of making use of the latent heat of evaporation and condensation of a refrigerant fluid. This is illustrated in Fig. 5.

Most of the refrigerants in use today are halocarbons. This is the group of refrigerants which contains one or more of the halogens; fluorine, chlorine and bromine, and are commonly known as the Freons.

I do not propose, in this paper to enter into a long discussion of the suitability of the many available Freons, but the thermodynamic choice results from a compromise between a low freezing point and a high critical temperature.

After considering all the other physical properties, such as toxicity, water absorption, etc., the best "compromise" choice appears to be Freon 12. However, this refrigerant has a critical temperature of 112°C, which means, in practical terms, that the heat sink upper temperature limit is between 65°C and 70°C, in order to provide reasonable system performance. This is an obvious limitation in a military aircraft where ram temperatures up to 150°C can be experienced.

The other big disadvantage of vapour cycle systems in the small miliary aircraft is the weight and bulk of the necessary equipment. The weight penalty would be about four times that of the equivalent capacity air cycle system.

The conclusion is that, in its present form, vapour cycle refrigeration is not really practical for the high performance military aircraft.

3.3 Air Cycle Refrigeration (See Fig. 6).

Air cycle refrigeration is the technique which has been predominant in aircraft environmental control ever since air conditioning and refrigeration became a necessary and recognised part of an aircraft.

There are many variations on the basic cycle which can be used, but the basic principle remains the same. The fundamental concept is to provide a source of fairly high pressure air, reject as much heat directly to ram air (or fuel) as temperature levels allow, and achieve the final cooling by expanding the air through a turbine, acting on some form of brake. In most cases the air supply comes from tapping the engine compressor, although in some cases, separate mechanically driven compressors have been used.

Scheme 1 shows the simple cycle, or turbofan system. In this cycle, the charge air is reduced in temperature by passing through an ambient (ram) air cooled heat exchanger, and then expanded through a turbine, the power of which is absorbed by a fan, used to assist the cooling air flow.

This type of system is mainly used in "low" performance sub-sonic aircraft, which do not experience high ram air temperatures.

Scheme 2 shows a conventional bootstrap system. In this cycle, the turbine drives a compressor which increases the pressure at the turbine inlet. In doing so, the temperature of the air is increased, which allows more heat rejection to ram air. The increased turbine pressure ratio provides a greater temperature drop across the turbine which is essential for providing adequate cooling in high ram temperature flight conditions. This type of system is frequently used in higher performance aircraft.

Scheme 3 shows a regenerative reversed bootstrap system. This type of system is effectively an air cycle heat pump. Apart from the initial precooling, (generally to ram air) no additional external heat sink is required.

After pre-cooling, which cools the charge air down to just above ram temperature, passing through a regenerative heat exchanger and expansion turbine cools this charge air to a temperature which can readily be used for extracting heat from a secondary cooling medium by means of another heat exchanger. The turbine drives the bootstrap compressor which in this case reduces the turbine outlet pressure, increasing the turbine pressure ratio and, of course, its temperature drop. From the compressor, the air is discharged overboard. This type of system is more suitable for closed loop cooling systems, for example as used in some radar and E.C.M. transmitters.

4. HEAT TRANSFER AND TRANSPORTATION TECHNIQUES (GENERAL)

4.1 Direct or Indirect Cooling?

Having generated a suitable temperature heat sink by using refrigeration, a decision has to be made whether the heat from the heat source should be taken to this heat sink, or the heat sink taken to the heat source.

In the former of these two alternatives, the heat from the source is transferred to a heat transport medium, and taken to a different part of the aircraft. (Indirect cooling).

In the latter, the heat sink (generally cooled air) is taken directly to the heat source (Direct cooling).

4.2 Direct Cooling

Taking cooled air to the heat source is self explanatory and is done in many cases. Cabin conditioning is the obvious example. Cooled air is blown directly into the cabin and then discharged overboard, or ducted to its next use.

Cooling avionics by blowing air directly through (or around) black boxes is another example of this method.

What is wrong with this technique?

In general, it is probably the most inefficient means of using expensive refrigerated air that one can imagine, even though it is extensively used.

Fig. 7 illustrates the disadvantages and inefficiencies:-

(a) Ducting air through aircraft necessitates the use of large diameter pipes which, in the small compact aircraft, are difficult to install.

(b) Even with lagging, the effectiveness of the cooled air is reduced by heat pick up.

(c) Blowing air directly at the heat source is not efficient in cooling the parts which really need cooling and frequently means large losses of cooling air in order to direct sufficient air at the required spot.

(d) Indiscriminate use of cooling air in this way frequently provides cooling to parts which do not really need it, thereby increasing the total heat load.

(e) It is very difficult to collect the cooling air for further use (e.g. in regenerative systems).

Despite the long standing traditional concept of cabin air conditioning, which will be very difficult to change, we must conclude that there are better ways of cooling which can, and must be employed wherever possible.

4.3 Indirect Cooling

Indirect cooling, or taking the heat to the heat sink can be a more efficient method, even when only used partially. The prime advantage is that it takes heat (or should do if well designed) from the parts which actually need cooling and not from the surrounding ambient.

However, effective use necessitates very good heat transfer at all system interfaces. Otherwise, temperature gradients can occur between the source and sink and to a large extent negate any other benefits.

There are many ways in which the principle can be applied:

Fig. 8 shows three simple examples of them.

In example 1, the liquid cooled suit (L.C.S.) enables the man's deep body temperature to be kept within comfort limits in hotter ambient temperatures.

In example 2, a liquid cooling loop is used for direct cooling of a radar transmitter, and a secondary liquid loop is used (via a liquid/liquid heat exchanger) to transport the heat to the air cycle refrigeration system, which may then be situated in another zone of the aircraft.

Example 3 shows in a simplified manner, how the principle is applied to avionic box cooling whereby the heat generated by the components is conducted (by metal strips) to the "cold wall". The cold wall can then be cooled either directly by air, or indirectly by a liquid cooling loop.

In the examples quoted above, the liquids most likely to be used are water/glycol for the L.C.S., and one of the Coolanols for the radar or electronic equipment.

The above techniques are readily available and do not require any advancement of technology for their use.

A method of heat movement which needs further development, but may prove to be an important part of tomorrow's heat transfer systems, is the heat pipe.

Figure 9 illustrates the principle of the heat pipe, which makes use of the latent heat absorbed and given up during the process of changing state from liquid to gas or vice versa. Heat from the source evaporates the heat pipe liquid, which then flows in gaseous form to the heat exchanger of the heat sink, where the gas is condensed to liquid. The liquid is carried back to the heat source by means of a wick. In this way, it is possible to conduct far more heat per unit area than other more conventional means. The choice of fluid depends on the design temperature levels of the source and sink.

Heat pipes are in use today in some fixed installations in industry and also in space technology. In these conditions, there is either a fixed height difference between source and sink, or, in the case of satelites, the environment is non-gravitational. Before such devices can be used in the military aircraft, further development of suitable wicks is required, in order to provide adequate liquid flow in conditions of varying 'g' and buffet.

5. APPLICATION OF COOLING TECHNIQUES IN AVIONIC AND ELECTRICAL EQUIPMENT INSTALLATIONS IN AIRCRAFT

5.1 So far, this paper has dealt with general principles of cooling which can be used in designing aircraft systems.

Let us now look at practical applications of these heat transfer techniques for cooling avionic and electrical equipment in military aircraft.

5.2 Air Cooled Black Boxes

The equipment bays of most military aircraft contain equipment which can be either "off the shelf" (meaning it was initially designed for another application) or designed specifically for the current aircraft. Because of this, it is frequently necessary to install equipment using different cooling methods, side by side.

For the purpose of cooling, the equipment can be divided into four types:-

(i) Equipment relying on radiation and convection from the outer case for its cooling;

(ii) Equipment whose heat loss is assisted by a supply of airflow blown through the box from the aircraft supply system;

(iii) Equipment which is cooled by a supply of air from the aircraft system blown through a "cold wall"/heat exchanger;

(iv) Equipment which uses fans to induce a supply of cooling air from the surrounding ambient. This fan induced air is then used for either direct or indirect component cooling in the black box.

At first sight, the first of these four types may seem attractive from the point of view of cooling airflow demands. This would be the case if the components within the box were capable of operating at high temperature and still give satisfactory reliability. Unfortunately, this situation does not often arise, and the majority of these "uncooled" boxes, when installed in an aircraft equipment bay require more cooling airflow blown "around" the box in order to control the local environment than many well designed force cooled items.

Hence, for all but the very minor items of equipment, the "natural convection and radiation" technique is not favoured.

The second of the four types, (i.e. simple forced cooling) is a natural step to try and improve the component temperature level from that existing for the first type. In too many cases, this natural step has led to some of the most inefficient air cooling methods, with excessive demands for cooling air not achieving the desired effect of reduced component temperature.

Of the four types listed the third ("cold-wall" heat exchanger) can be the most efficient, if well designed. Emphasis must be placed on the requirement for good thermal design, playing a significant part in the box layout. Without these considerations, the potential of this type of design will not be realised in practice.

In closely packed equipment bays, the use of fan cooled equipment is not favoured. This can lead to re-circulation problems within the bay and so it is generally better practice to supply any forced cooling requirements from the aircraft air supply system.

It is common practice to install such equipment on mounting trays using ARINC standards for dimensions, 'hold down' methods, etc. The mounting tray can then be used for the air distribution system.

Figure 10 shows how black boxes of different cooling standards can be accommodated side by side using this installation method.

In the more complex aircraft, employing many items of equipment, airflow distribution can become quite a problem. Where airflow is limited, it cannot be wasted, and therefore accuracy of flow distribution becomes essential. Accurate control of pressure drop in each box system is necessary during production, to ensure satisfactory operation.

5.3 Liquid Cooled Equipment

Until recently, liquid cooling of black boxes has not been in common use, although some equipment (including radar) has used liquid cooling for several years. In some American aircraft, liquid cooled avionics packages are becoming more common.

As mentioned previously, the liquid is purely a heat transfer and transport medium. The most frequently used liquid is Coolanol 25.

A typical liquid cooling loop is shown in Figure 11. It can be seen that the main items in the system are heat exchangers, a liquid pump and an expansion chamber.

In most cases, the mass of the liquid and its associated equipment is less than that required to do a similar job using all air cooling. An improved cooling potential, together with more design flexibility, stand out as distict advantages. NOTE: Improved cooling efficiency can result from two main aspects:-

(i) Cooling of the heat source only and not the surrounding ambient;

(ii) Lower thermal gradients within the equipment due to higher heat transfer coefficients, and higher specific heat.

However, let us not pretend that a change to liquid cooling for general avionic equipment would be an easy or inexpensive step to take.

Before any such method could be adopted for future aircraft designs, a great deal of investigation is required into the many aspects apart from cooling.

For example, it is probably not very practical to consider breaking down liquid cooling pipes during routine equipment maintenance, with the inevitable mess resulting from liquid spillage. A "sealed" liquid loop, being part of the aircraft, would probably be required.

Great care must be taken to choose the most suitable method of cooling for the job in hand. Examples have been known where equipment designers have advocated liquid cooling for small heat loads (the largest portion being kinetic) where insulation and a small air supply provides a cheaper and more effective solution.

6. OVERALL CONSIDERATIONS AND CONCLUSIONS

This paper has illustrated but a few of the techniques which are currently available to the aircraft and equipment designer.

From these techniques only, and without any revolutionary scientific breakthrough in cooling methods, it is possible to achieve a much higher cooling efficiency and lower aircraft penalty, simply by ensuring a closer integration of equipment and aircraft design than has been achieved in past aircraft projects.

For future aircraft, it is essential that the base for this integration should start well ahead of the actual aircraft project design.

In order to achieve anything like the full potential improvement in avionic equipment cooling, we must consider the complete system, as installed in an aircraft, resulting in a New Installation Concept for Military Avionics.

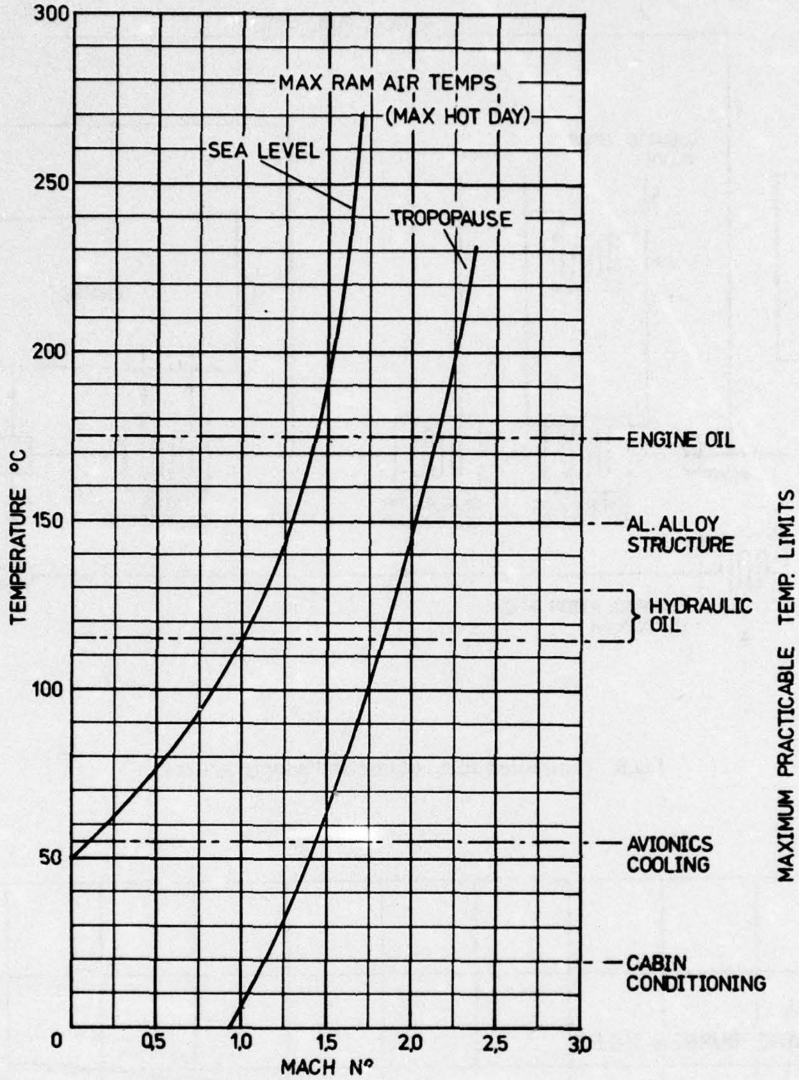


Fig.1 Maximum temperatures of ram air and temperature limits

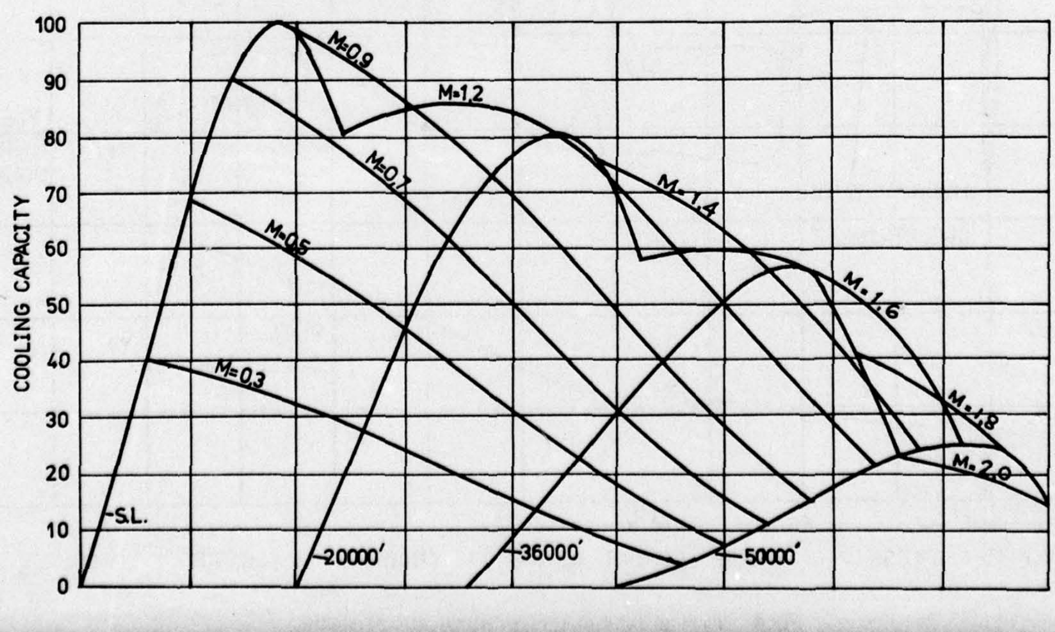


Fig.2 Variation of cooling capacity with altitude and Mach No

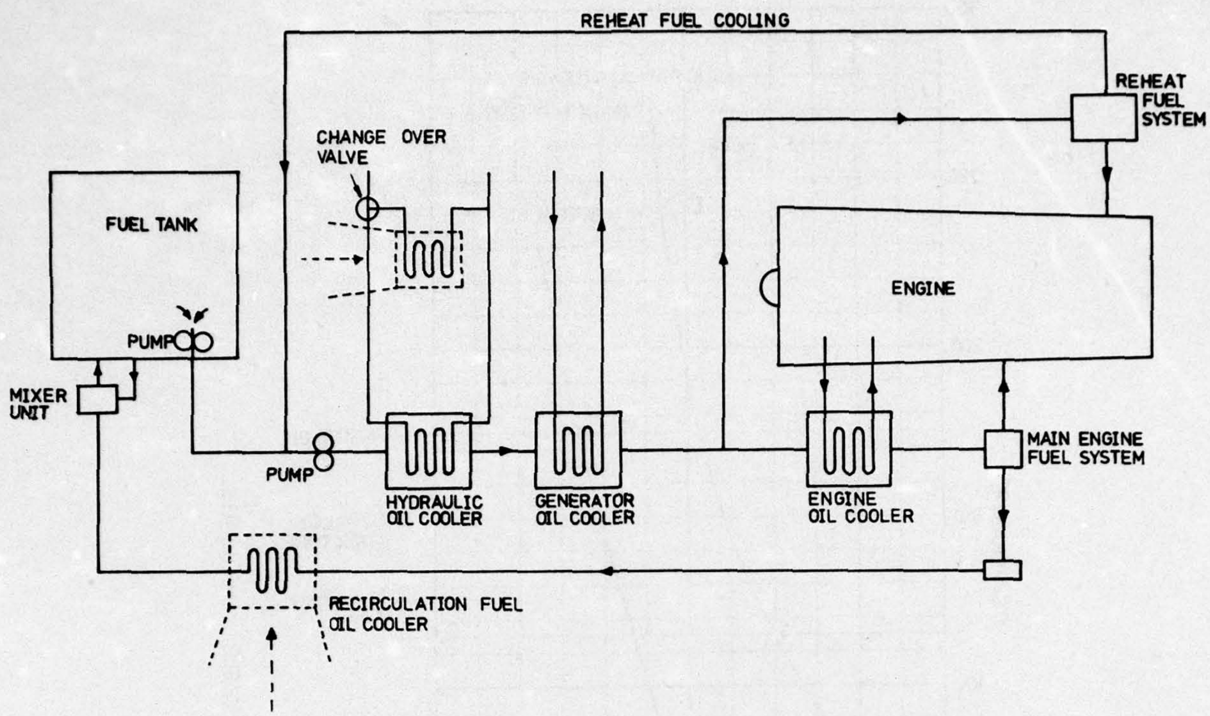


Fig.3 Simplified fuel cooling and supply system

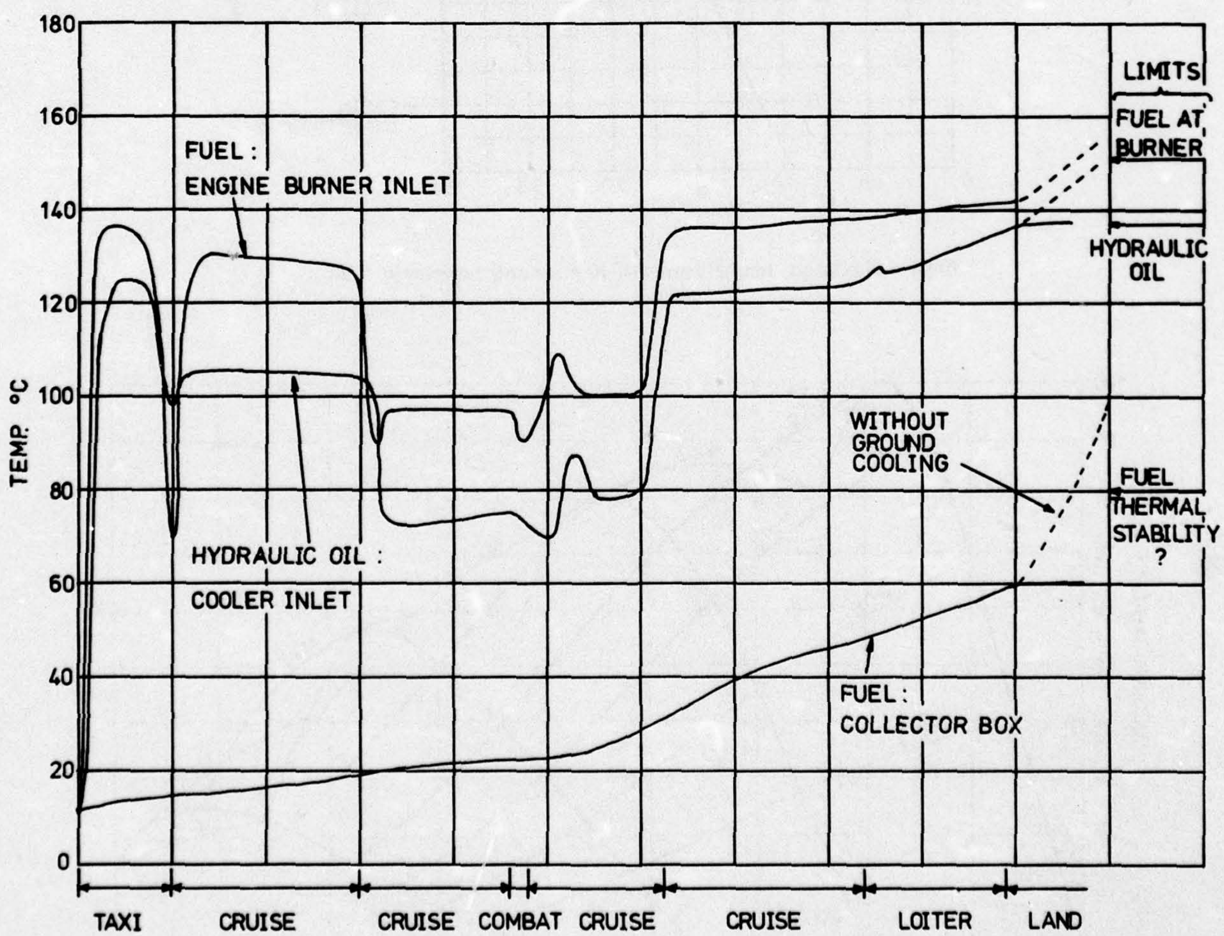
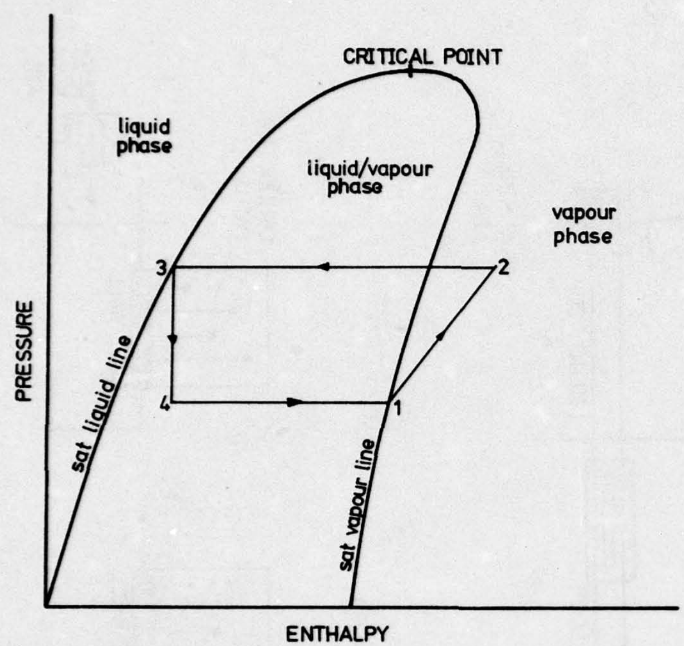


Fig.4 Typical system temperatures during sortie



1–2 ISENTROPIC COMPRESSION OF VAPOUR RAISES TEMP. AND PRESSURE

2–3 VAPOUR REJECTS HEAT AT CONSTANT PRESSURE IN THE CONDENSER AND ENTERS THE LIQUID / VAPOUR REGION

3–4 FLUID IS EXPANDED ISENTROPICALLY TO ITS ORIGINAL PRESSURE

4–1 LIQUID REFRIGERANT IS RE EVAPORATED AT CONSTANT PRESSURE BY HEAT TAKEN IN FROM L.C.S. SYSTEM

Fig.5 Simplified vapour cycle

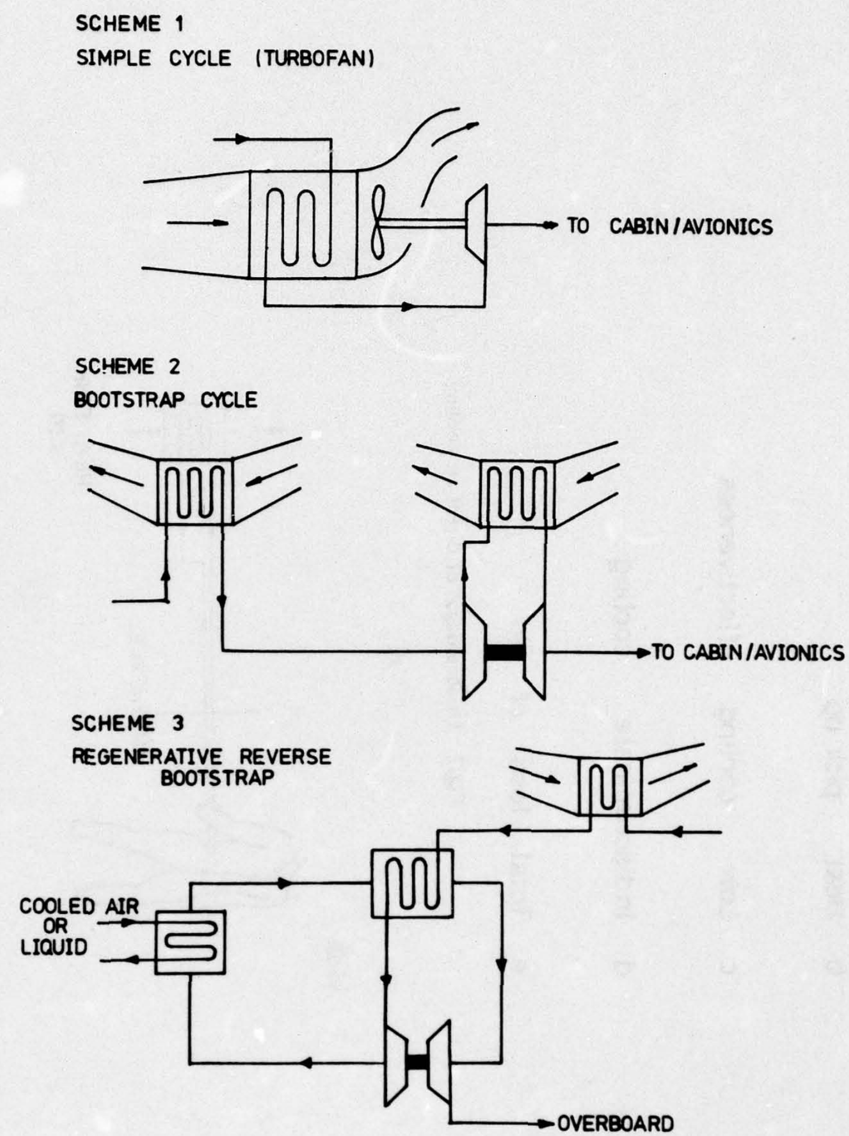


Fig.6 Examples of air cycle refrigeration

a. Large diameter pipes - installation difficult.

b. Heat pick up.

c. Low cooling effectiveness.

d. Indiscriminate cooling.

e. Total loss of air.

Fig.7 Disadvantages of direct air cooling

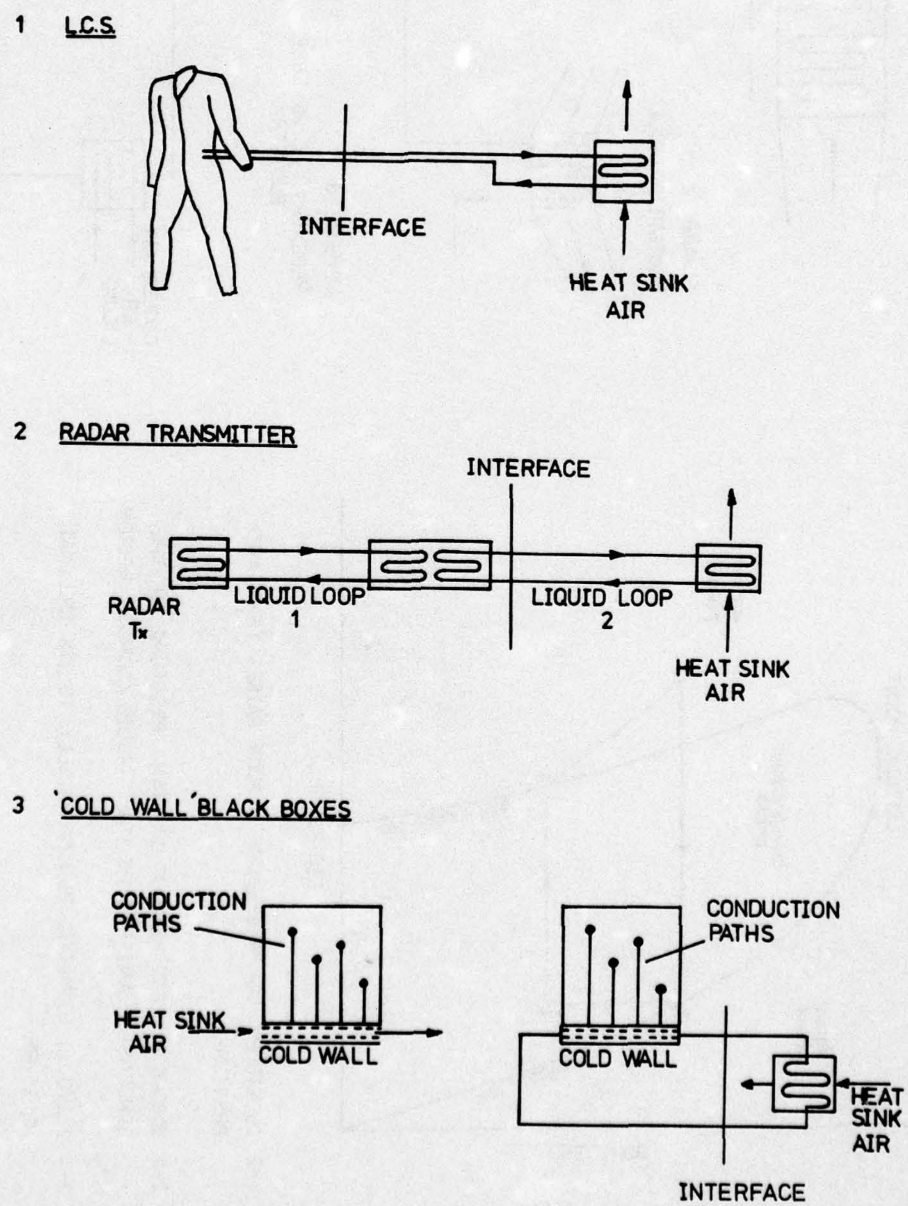


Fig.8 Examples of indirect cooling

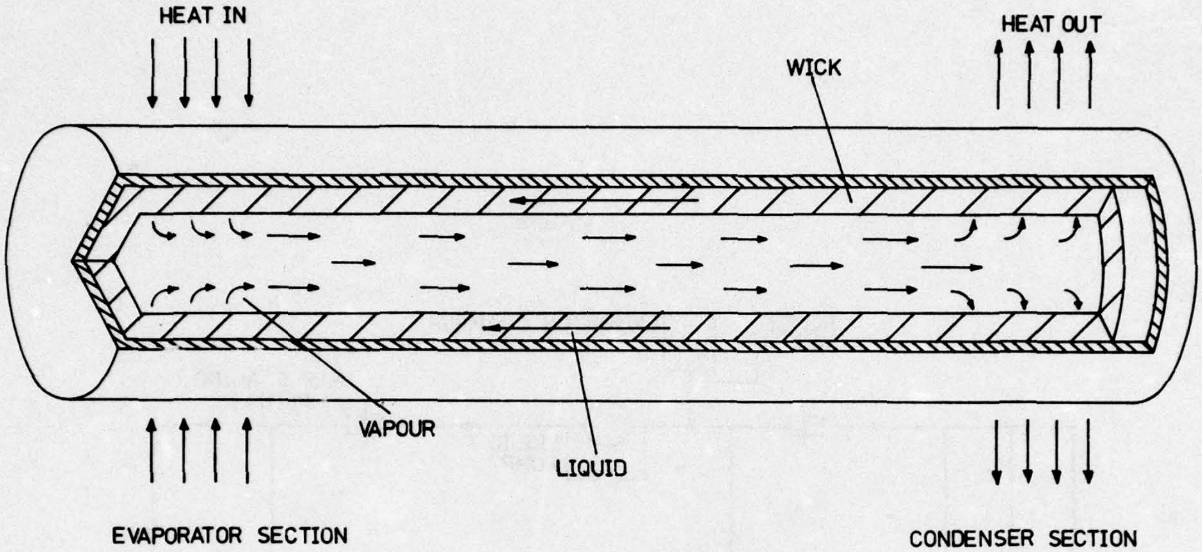


Fig.9 A heat pipe

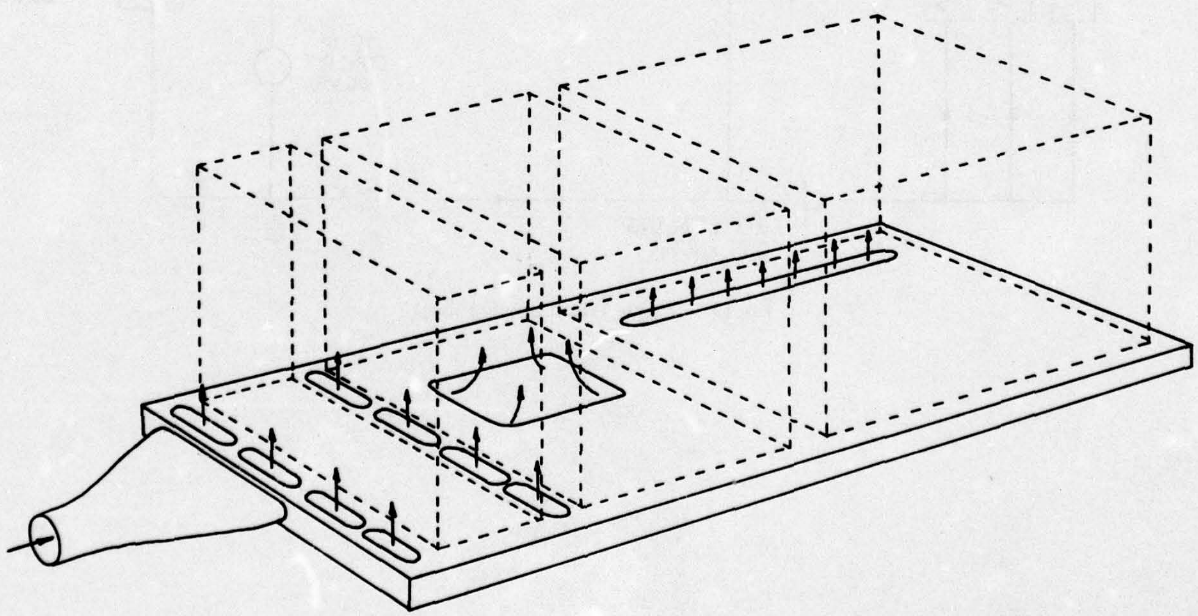


Fig.10 Typical cooling tray

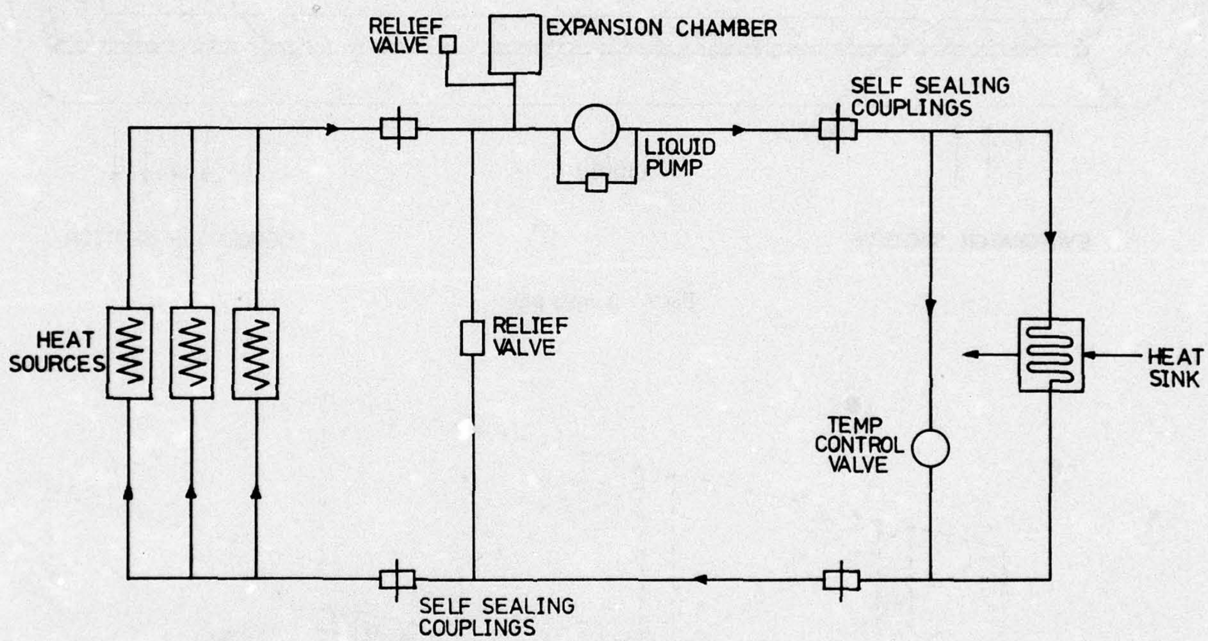


Fig.11 Simple liquid cooling loop

DISCUSSION

K D Groom:

Have you any reservations about forced cooling through black boxes which contain mechanical movements. Can you cite specific problems of knowledge of same.

K Morgan, BAC Warton

A leading question! The answer would seem to hinge upon the quality of the incoming forced cooling air. Dust and moisture content in the cooling supply depends obviously on system design and to some extent on whether external air supplies are drawn in for say ground cooling. If pure dry air were guaranteed, then direct forced cooling through black boxes containing mechanical movements would seem acceptable. However, as this cannot normally be shown to be the case then direct forced cooling would be inadvisable.

I cannot cite specific examples where direct forced cooling of such boxes has caused problems but I have noticed an awareness among some equipment manufacturers of the contamination aspects discussed above. This has resulted in significant movement towards cold wall cooling in my current activity in the Military Aircraft field.

THERMAL MANAGEMENT OF FLIGHT DECK INSTRUMENTS

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and
G. W. Brooks

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SUMMARY

Screening tests, which are part of a Boeing research program for improving avionic equipment cooling, were conducted to identify limits and deficiencies in current crew station instrument panel cooling systems and to investigate advanced cooling systems which extend or remove these limits. A test article simulating a 16 unit, engine instrument panel was constructed. The advanced concepts have cooling designed into the instrument panel structure; one includes a simplified retaining method for the units. Significant results for the baseline concept (current) are cascading temperature effects and thermal sensitivity to the uncontrolled airspaces behind the units which limit them to low power units. The advanced concepts exhibited significantly improved and predictable cooling performance with lower and uniform surface temperatures and at power levels well beyond the baseline concepts. Cascading effects and geometrical sensitivity were absent in the advanced concepts. The advanced concepts have potential payoffs in improved avionic equipment reliability and crew performance which will reduce life cycle cost, ensure mission completion and improve safety.

INTRODUCTION

The design trend in Aircraft Avionic Systems toward the use of lightweight high-power density units has necessitated a new appraisal of cooling systems for avionic equipment.

Increasing operating temperatures of avionic equipment in the crew station has been the impact of this trend with a resulting reduction in reliability and crew performance. The importance and current interest in temperature effects on reliability is evidenced by ARINC408A, 1975 and Dantowitz, 1971. This interest is also shared by Boeing. If the temperature levels of the components within these units can be uniformly and adequately controlled, the resultant improvements in reliability and crew performance will reduce life cycle cost, ensure mission completion and improve safety. The thermal control of components within a unit is dependent upon the supplier's internal thermal design as well as the airframe manufacturer's cooling system design.

Thermally related reliability can be difficult to isolate; for example, a review of a group of five different instruments indicated a mean time between unscheduled removals of 230 to 1600 hrs, while the specifications for these units called for a mean time between failure from 4500 to 6000 hr Failures which could be verified were in the order of 30 to 40% with evidence that a high percentage was thermally related, such as, capacitor and motor bearing failures. This type of data indicates that the potential for improved reliability can be considerable and is reflected by recent activity in updating industry standards. Current standards which can affect the design of instrument cooling systems are included in U.S. Military Specification MIL-E-38453A, Environmental Control, Environmental Protection, and Engine Bleed Air Systems, Aircraft, General Specification for, 1971 and the proposed Specification No. 408A Air Transport Indicator Cases and Mounting, 1975. MIL-E-38453A, 1971, allows a 105°F (40.6°C) touch temperature with an 80°F (26.7°C) ground ambient. This can be compared to the proposed ARINC 408A change which allows a unit's front surface to be at 130°F (54.4°C), the unit's case at 150°F (65.6°C) when the supply (e.g. crew station ambient) is 100°F (37.8°C) and the coolant airflow rate is based on 8 lbs/(Min-Kw) (3.6 Kg/(Min-Kw).

Boeing has initiated programs, which include a recently completed cooling concepts screening test, whose purposes are to identify limits and deficiencies of existing crew station instrument-panel cooling systems and to determine how these limits can be extended or removed by more efficient cooling system designs.

Acknowledgements: The authors express their gratitude for encouragement given throughout the program by Messrs. W. G. Nelson, Unit Chief of Mechanical/Electrical Systems Technology Research and Narinder S. Attri, Chief Engineer of Boeing Military Airplane Development organization (BMAD), Mechanical/Electrical Systems Technology. Special thanks go to Messrs. A. S. Yorozu and R. L. Slack for their interest and contributions which continue to make the avionic equipment cooling program a success.

The overall program for the crew station cooling investigation was designed to concentrate on two distinct problem areas. These are the individual unit and the complete instrument-panel installation, including cooling ducts, etc. The program includes:

1. Evaluate current systems to identify problems.

2. Analyze flight hardware to identify typical detail thermal design problems with emphasis on a single unit and design advanced systems to overcome current problems.

3. Using 1 and 2 for guidance, conduct cooling concepts screening tests, simulating both current and advanced systems with the objectives of reducing temperatures and defining limits.

4. Based on results of 1, 2 and 3 conduct analysis and tests on complete installations to investigate design problems using actual hardware.

5. Improve internal thermal design consistent with advanced concept's performance.

The purpose of this paper is to present the results of the above screening tests. Prior to these tests and, as a general assessment of the thermal design problem, it was essential to review current cooling systems to determine the potential payoffs in reliability and to investigate the order of magnitude of temperatures and heat loads in the crew station instrument-panel installation.

CURRENT SYSTEMS

Figure 1 illustrates an E-4 (Model 747) command post airplane crew station instrument panel installation. The units are typically spaced 1/4 in. (.635 cm) apart and are not uniform in cross section and length. Figure 2 illustrates the model 747 cooling system which is described as a push-pull system where crew station air is drawn beneath the units from cabin air and supplied by an outlet under the units when the air conditioning packs are on. Air removal is accomplished by a general evacuation system. Other Boeing models such as the T43A (Model 737), have similar cooling concepts except air is evacuated at the top of the units. The maximum unit heat load in these systems is in the order of 0.3w/in^2 (.047 w/cm^2) based on an instrument case surface area.

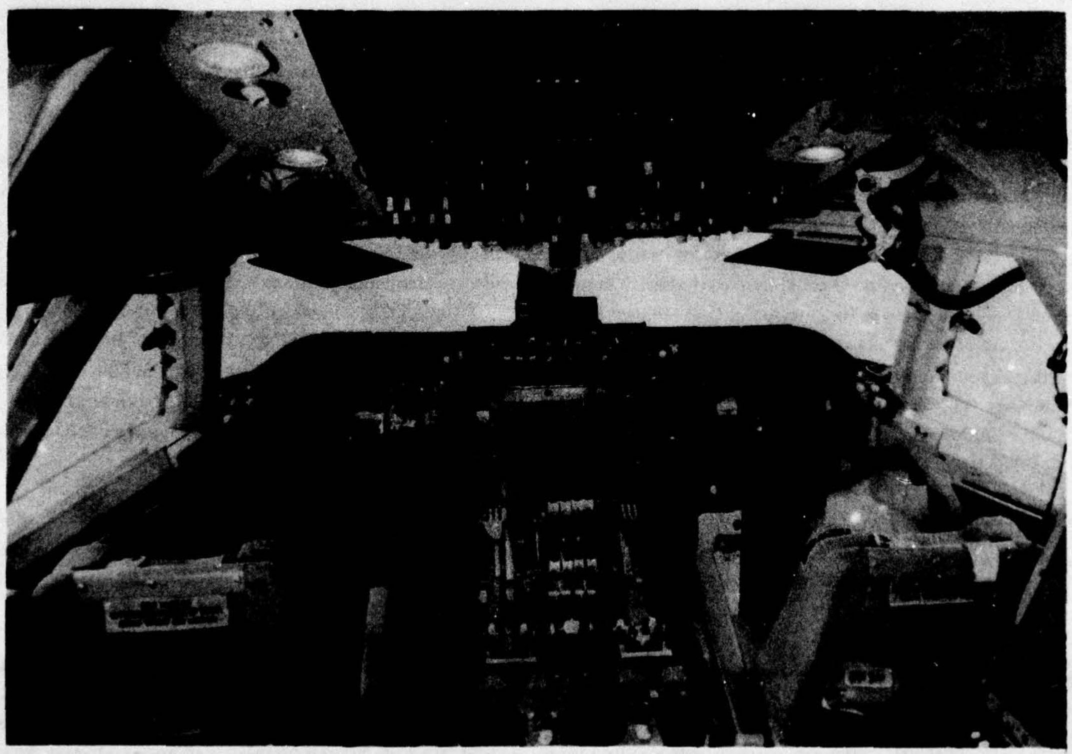


Figure 1.-Model E-4 Crew Station Instrument Panel Installation

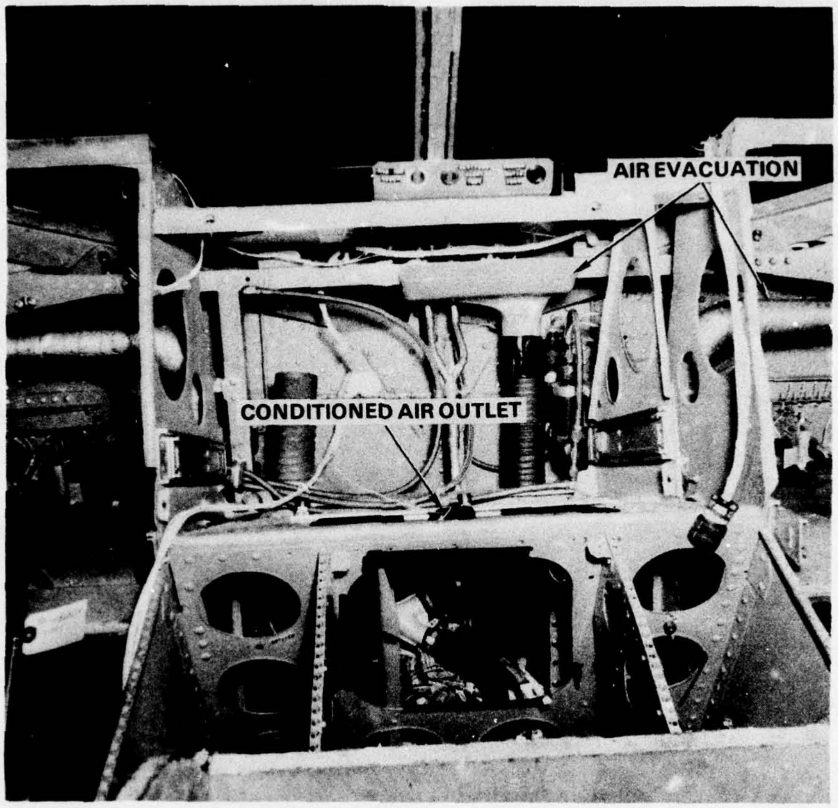


Figure 2.–Model E-4 Crew Station Instrument Cooling Installation

ANALYSIS

A computer thermal analysis of a prototype unit was conducted to determine a typical unit's heat load and temperature distribution. For the analysis a prototype vertical scale unit was chosen, since it represented a modern design.

The unit has a 1.75 in. (4.44 cm) by 6 in. (15. cm) frontal surface and is 9 in. (22.8 cm) long with a heat-flux of 0.1 w/in^2 (.016 w/cm^2). The design conditions are consistent with the proposed specification ARINC 408A, 1975. Forced convection conditions were assumed with airflow directed vertically upward with local airflow distribution at the sides and back determined by the percentage of flow area available. In an actual installation, mixed free and forced convection conditions could exist. However, at the low velocity conditions, dictated by the low power conditions, both free and forced heat transfer coefficient can be of the same order and the assumption of forced convection is adequate.

Typical results from the analysis are indicated in figure 3. Significant conclusions drawn from the analysis are as follows:

- Backspacing behind units is an important design parameter.
- The side surfaces have the largest surface areas (67%) but were not being used effectively (30% of heat load).
- The front surface is one of the smallest surfaces (10%) but is rejecting a large percentage of heat (35%) back (reflux) to the crew station.
- Case temperatures are uniform due to the high conductivity of the case.
- Internal and external temperature control is important (i.e. ΔT internal = 2 x ΔT external).
- The bezel and glass temperatures are important design parameters as shown in figure 4.

Figure 4 is a plot of threshold of pain (touch) temperature for various materials. This type of data shows the importance of maintaining low-front surface temperatures and is part of the basis for design criteria such as ARINC 408A, 1975 and MIL-E-38453-A, 1971. Other considerations in the above criteria are the radiant heat effect on the crew and reflux of the unit's heat load.

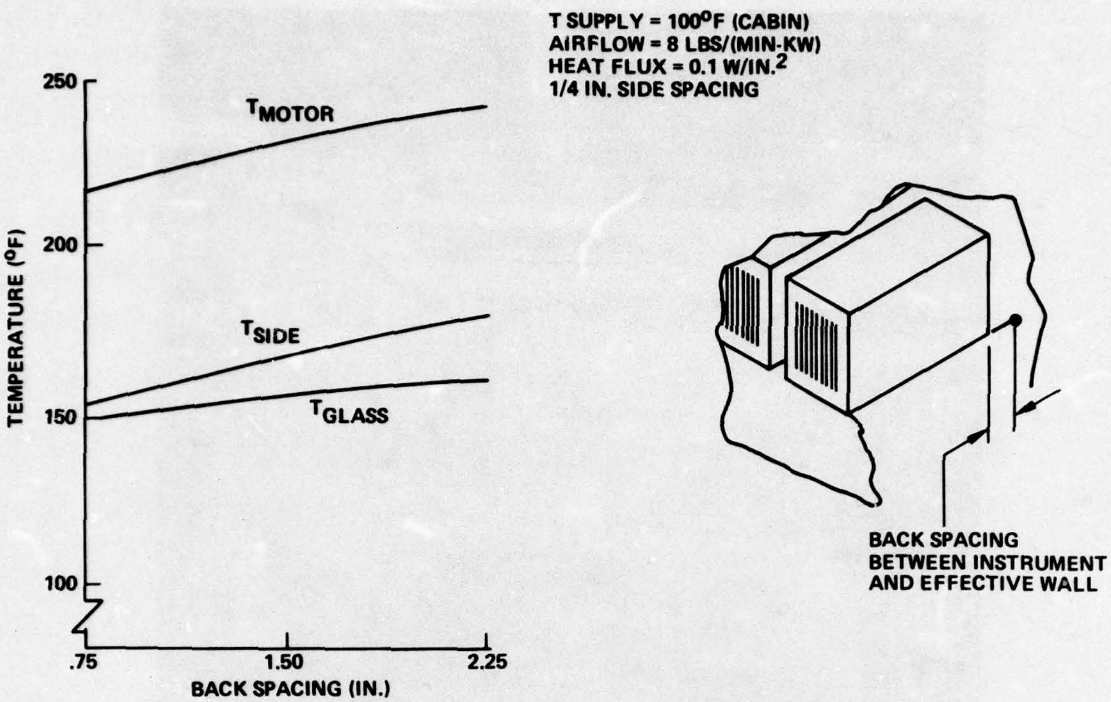


Figure 3.-Analytical Prediction of Forced Air Cooled Instruments

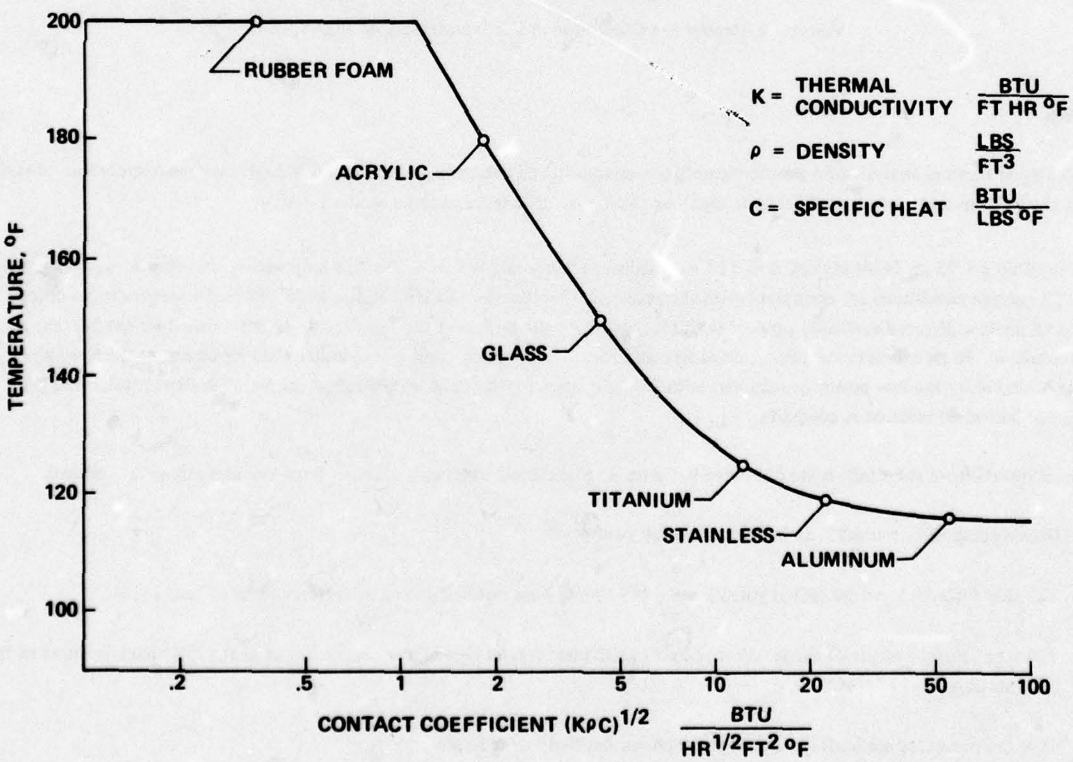


Figure 4.-Threshold of Pain Temperatures for Materials

DESIGN

Two types of advanced concepts were designed and designated as the suction and pressurized concept (see fig. 5). The suction concept is simpler to design and best applied to installations whose structure forms a complete enclosure. Its disadvantage is that, being a suction only system, its thermal effectiveness is hampered by leakages. The pressurized-panel concept provides integral clamping and mounting of a unit and is very tolerant of leakages since it is a push-pull concept. This latter concept is best applied to the pilot, co-pilot and engine instrument panels where instruments are not entirely enclosed.

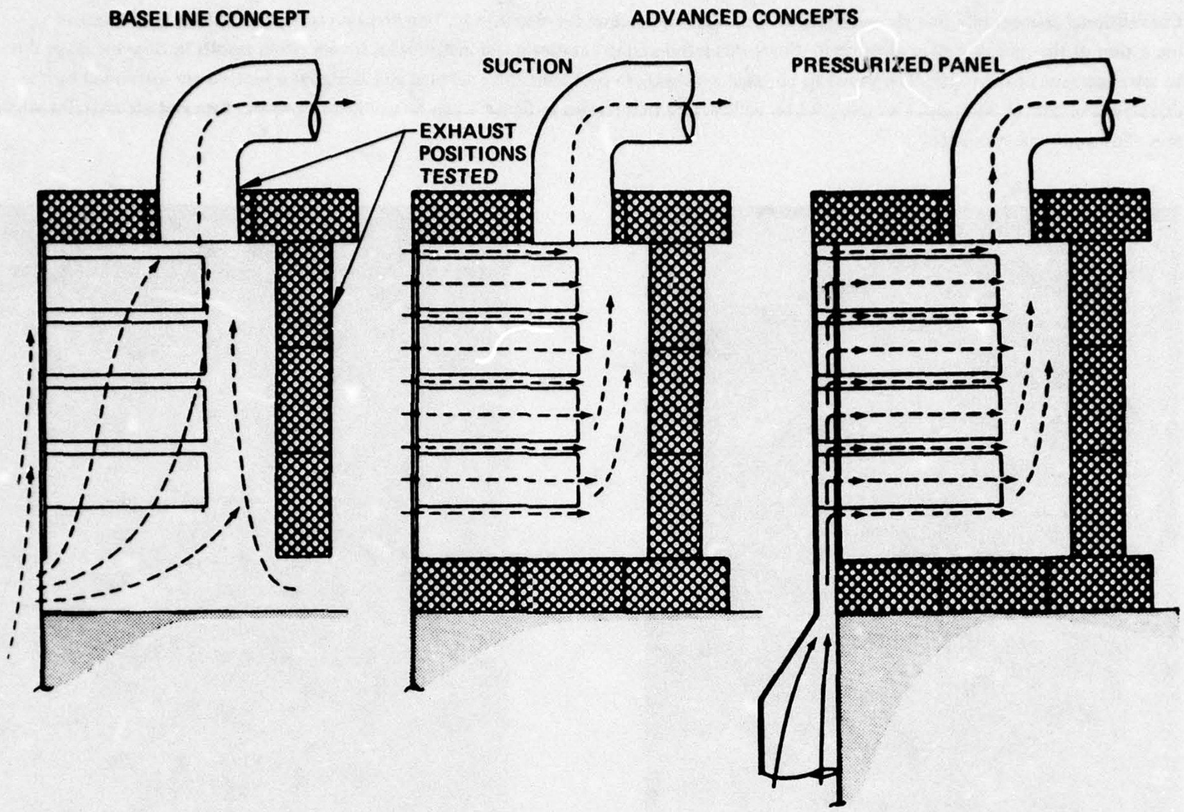


Figure 5.-Cooling Concepts Tested

Factors considered in selecting the design of the advanced cooling concepts include:

1. Maximum surface area exposure to coolant airflow
2. Touch temperature of the panel
3. Clusters of instruments and different power densities
4. Blockage and/or shortcircuiting of airflow by structure, wire bundles, switches, etc.
5. Different cross section, spacing, and/or length of instruments
6. Use of existing structure

In design of the advanced systems the first three factors dictate horizontal flow along the sides, flow from front to back, and control of airflow quantity by adjusting orifice size and/or quantity. The remaining factors were satisfied by supplying air from the front around the unit's periphery.

Figure 6 illustrates the arrangement of the orifice or holes around the unit's cut-out in the front panel which is utilized in the suction panel shown and in the pressurized-panel concept. The 1/8 in. (.317 cm) dia holes allow the air to expand into the 1/4 in. (.635 cm) space between the units creating a nonuniform velocity profile which is intended to create turbulent flow with accompanying high heat transfer rates. In the suction concept the air is drawn from the laboratory ambient, through the 1/8 in. (.317 cm) dia holes (fig. 6), expands as a jet into the space between sides of the units and continues on a horizontal direction until it is evaculated by a general exhaust duct (see figure 5).

The pressurized panel concept, figures 7 and 8, utilizes the above suction panel which is assembled with a baseline panel and spaced 3/4 in. (1.91 cm) apart to form a plenum. The airflow for the pressurized panel is illustrated in figures 5 and 8. The supply air is delivered by a duct and enters the assembly at the edge of the plenum. It then passes vertically up the plenum and turns 90 degrees as it passes through the holes (fig. 6) in the same manner as the suction concept. The pressurized-panel can be used to simplify the instrument's retaining method. This is illustrated in figure 8 where the retaining method is shown integrated into the pressurized panel.

Conventional clamps, like that shown in figure 8 require two screws for clamp mounting and two screws to achieve the restraining action of the unit as well as alignment. Since this latter clamp cantilevers an instrument, it very often results in flow blockage due to misalignment of the units. The pressurized-panel avoids these problems since vertical and horizontal motion are restrained by the double panel and, in addition, a simple friction device, like that shown in figure 8 can be utilized to provide fore and aft restraint which also allows easy maintenance.

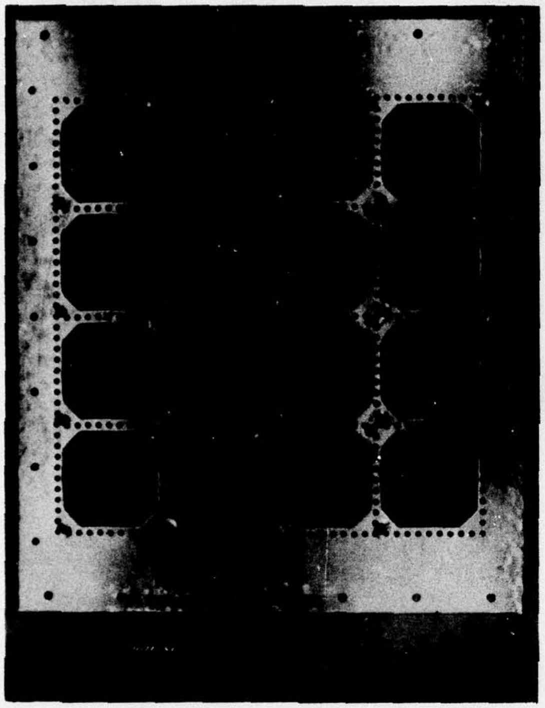


Figure 6.-Instrument Panel-Suction Concept

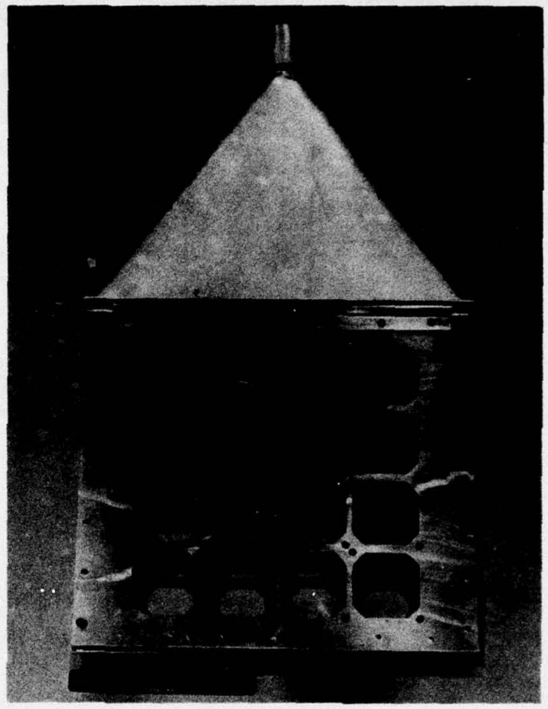


Figure 7.-Instrument Panel-Pressurized Panel Concept

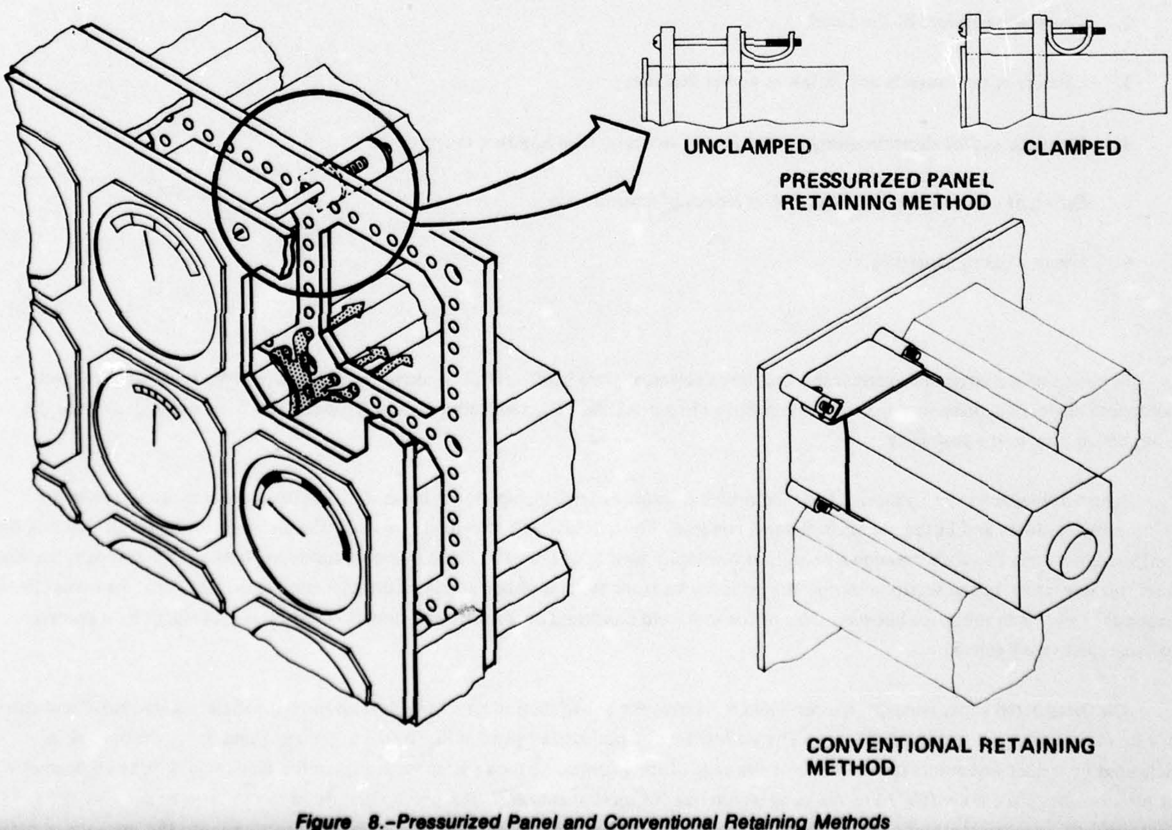


Figure 8.-Pressurized Panel and Conventional Retaining Methods

SCREENING TEST PROGRAM

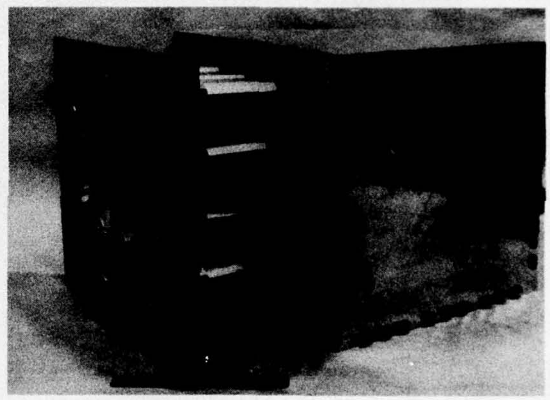
Using the analytical results as guidelines, screening tests were designed and conducted with the objectives of identifying, in detail, limits and deficiencies of current instrument-panel cooling systems and to investigate alternate and advanced cooling concepts.

The test article was a 16-unit, simulated engine instrument panel, typical of a multiengine aircraft, fabricated to determine, experimentally, the effects of variables encountered with an instrument panel cooling system. An engine instrument panel was simulated in this phase of testing because of its uniform arrangement and relative structural simplicity which aided in the test data interpretation.

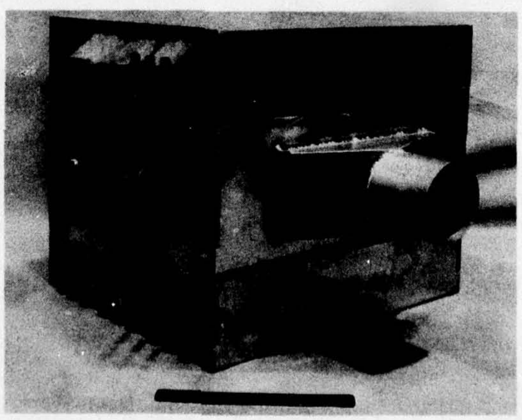
The three cooling concepts tested are shown in figure 5. The baseline concept is typical of current cooling systems. The other two systems are advanced cooling concepts designed to reduce the instrument case temperature and front panel touch temperature. Airflow patterns are indicated for the exhaust positions shown, while the exhaust positions tested are indicated.

Figure 9 illustrates the test article which was constructed to allow flexibility in testing and screening the cooling concepts. Supply and/or exhaust air positions, backspacing between the units and the test chamber rear wall, and side spacing of the units could be varied. The rear view of the test article in figure 9 indicates one of six possible exhaust positions. For example, the current systems mentioned previously (E-4 and T43A) could be simulated by drawing laboratory air from underneath the units and exhausting at the rear or top as appropriate. In addition, backspacing effects can be varied by positioning the rear wall assembly in one of three fore and aft positions as indicated by the nut plates in the vertical columns. Side spacing variations are accomplished by interchangeable front panels. The walls of the assembly were insulated with urethane foam to minimize heat losses. The heater assembly, figure 9, is an internally suspended resistance heater which provides uniform unit surface temperatures and variable and accurate power measurements. The simulated units utilize a standard 2.2 in. (5.59 cm) by 2.2 in. (5.59 cm) case by 6 in. (15.2 cm) in length (2AT1).

Heater Detail of a Simulated Instrument



Test Article with Simulated Instruments Installed



Rear View of Test Article— Multi-Positions Illustrated by Nut Plate Location

Figure 9.-Test Article for Instrument Panel Cooling Concepts

TEST SETUP

The test setup is illustrated in figure 10 with the pressurized panel installed. The front panel was instrumented with both heat flux meters and the thermocouples to determine the distribution of heat load in front and back of the panels. In addition, thermocouples were installed and positioned on the unit 3 in. (7.6 cm) aft of the front panel to coincide with the mid position of the 6 in. (15 cm)

long units in order to determine an average temperature of a unit. These were arranged on all sides of a unit. Additional thermocouples were installed in the air supply, ambient, exhaust and on the test assembly where heat balance data was required. A total of 70 thermocouples were utilized in the test setup.

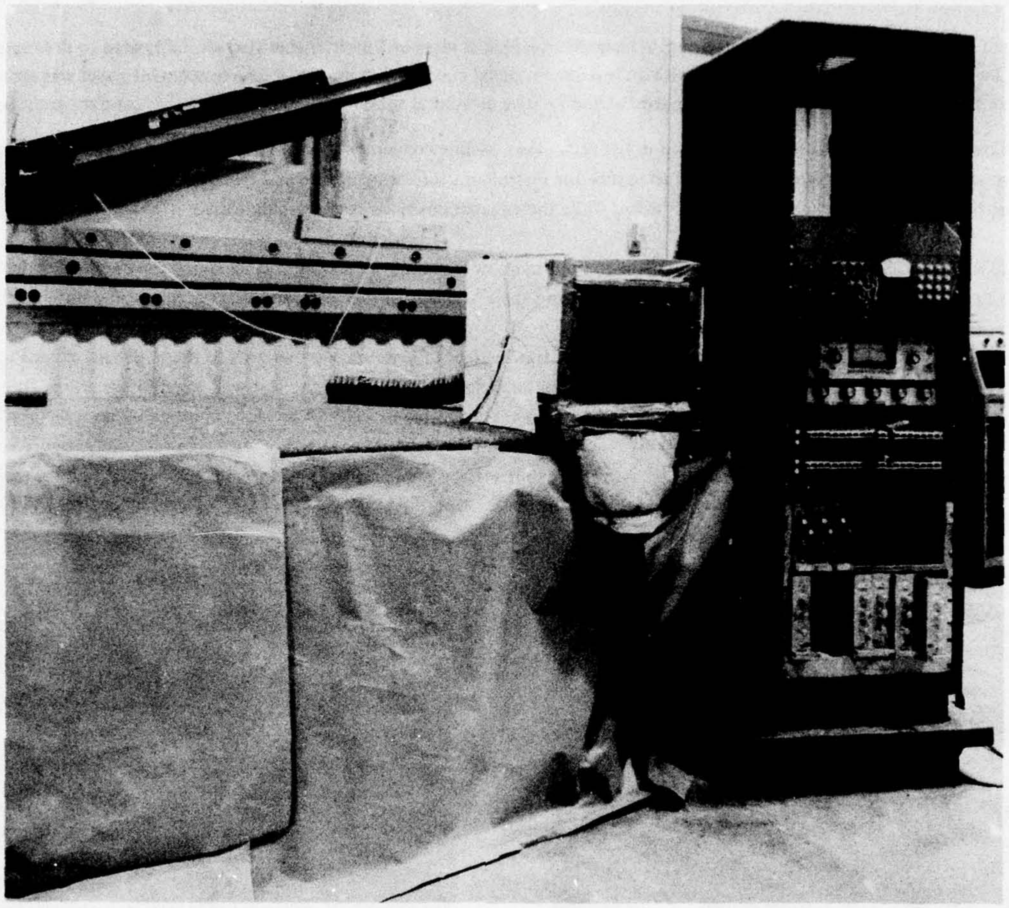


Figure 10.-Screening Test Set-up-Pressurized Panel

After reviewing current cooling system designs, future requirements and specifications, the scope of the screening tests was established as follows:

- Unit heat flux: 0.1 to 1 w/in^2 (.016 to 0.16 w/cm^2) of unit surface
- Coolant airflow: 8 to 20 lb/(min-kw) or 3.6 to 9.1 Kg/(min-kw)
- Backspacing to simulate effect of wire bundle space: 1.5 to 3.5 in. (3.8 to 8.9 cm)
- Exhaust duct position: top and rear (see fig. 5 for positions.)

A cutoff limit of 150°F (83.3°C) was established as the practical limit of surface to supply temperature differences in the testing.

TEST RESULTS

Two levels of data were desired from the testing. The first level was the average surface temperature of a unit above the supply air temperature (i.e. room ambient or pressurized-panel supply duct). This type of data allows a relative comparison between cooling concepts, geometry sensitivity evaluation and flow sensitivity comparison. The second level was the determination of heat transfer coefficients which were particularly desired on those concepts whose thermal behavior would be found predictable. This would allow empirical equations to be formulated which are useful to the thermal designer.

BASELINE TEST DATA

For convenience of the presentation the 16 units are numbered from left to right and top to bottom. The unit's average surface temperature above the supply (ambient) temperature for four of these units are shown in figure 11 and plotted against flow rate per kilowatt. These four units are selected since they illustrate extremes in the temperature data. The data is arranged so that two different heat-flux (w/in^2) levels can be compared on the left and right sides of the figure. Geometrical effects are presented as back exhaust and top exhaust in the upper and lower half of the figure respectively (see fig. 5 for actual location). In addition, the extremes in backspacing are shown on the left and right on the plots for each heat-flux level. Figure 11 shows that cascading temperature effects are one of the most dominant characteristics of the baseline concept where heat from one unit is passed to another in the upward airflow path. Cascading is noted to increase with both increasing heat-flux and backspacing as would be expected. The backspacing effect, which allows air to short circuit behind the units, can be observed by noting the temperature change of an individual unit when changing from 1.5 in. (3.8 cm) to 3.5 in. (8.9 cm) position. Exhaust duct position is the least sensitive geometrical factor for the heat-fluxes shown and appears to affect some of the units near the exhaust exit. It will also be noted that a given heat-flux, increasing flow is more effective at the higher heat-flux. However, increasing the heat-flux cannot be compensated for by a proportionate increase in flow.

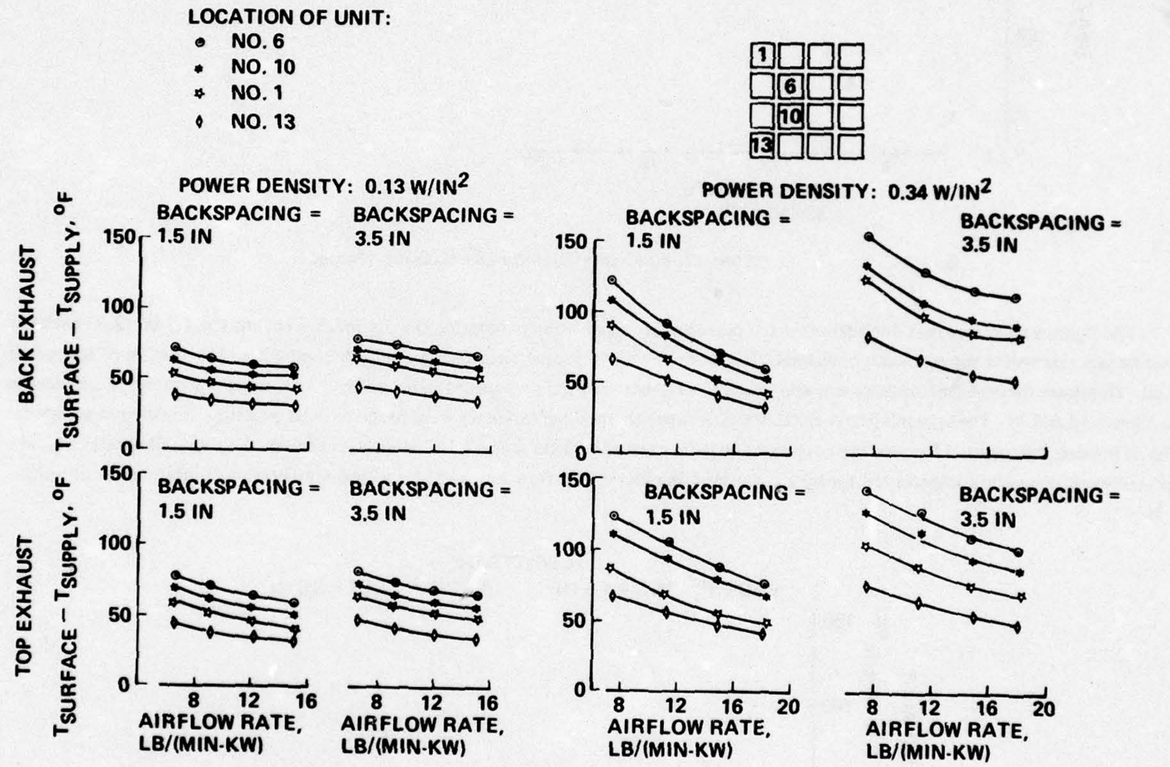


Figure 11.-Temperature Related to Heat Flux, Position, Flow and Geometry-Baseline Concept

Figure 12 illustrates the division of heat rejected from the front panel compared to that exhausted by the cooling system behind the panel. For example, at 8 lb/(min-kw) (3.6 Kg/(min-kw) and 0.13 w/in^2 (.020 w/cm^2), only 60 to 80% of the heat is being exhausted for the baseline concept. In an actual installation this would imply that 40 to 20% of the heat generated is being radiated and transferred by natural convection back to the crew station for possible reflux. The plots are, therefore, a measure of success of a cooling system. At higher heat-fluxes with higher flow rates, the increase in airflow velocities behind the panel results in higher heat transfer coefficients behind the panel which is reflected in a more favorable heat load distribution.

Tests were also conducted at 3/4 in. (1.91 cm) side spacing which shows a slight reduction of the cascading effect but no change in the general temperature level.

ADVANCED CONCEPTS TEST DATA

The advanced concepts exhibited significantly improved cooling performance coupled with lower surface temperatures and represented a substantial energy savings. This is illustrated for the pressurized panel in figure 13 for a heat flux of 0.34 w/in^2 (.053 w/cm^2). The absence of cascading temperature effects and the insensitivity to geometry changes is evident. The minimal differences in temperatures of center units (e.g. no. 6 or no. 10) compared to corner units (e.g. no. 1 or no. 13) is mostly due to the absence of heat transfer from adjacent units to the corner units.

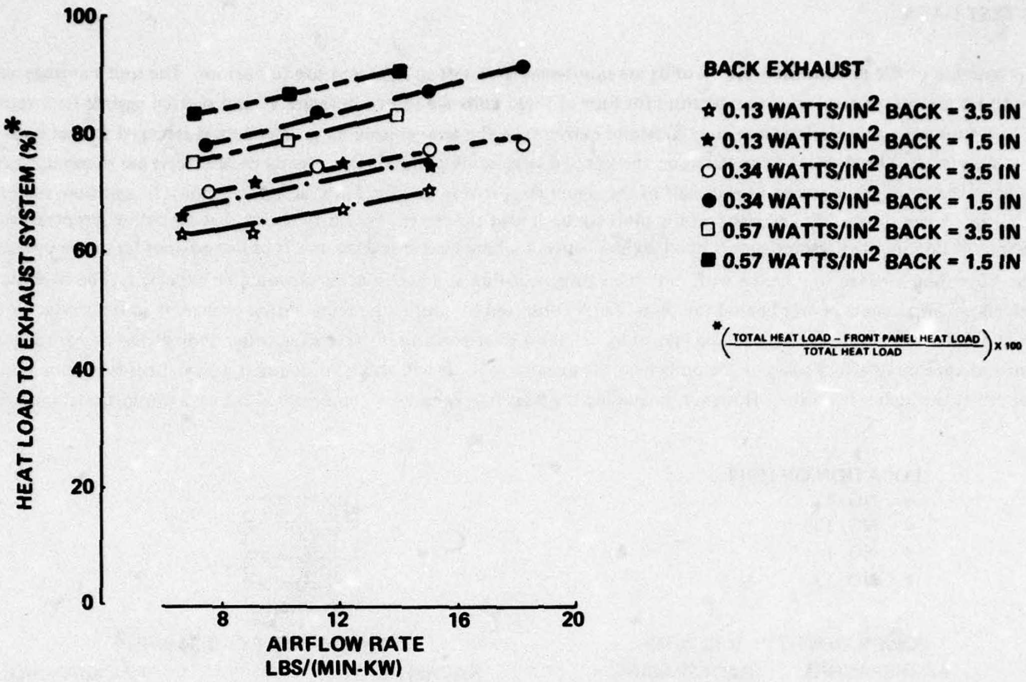


Figure 12.–Heat Load Distribution–Baseline Concept

The figure also shows that duplicate thermal performance exist when comparing the 3.5 in. (8.9 cm) to the 1.5 in. (3.8 cm) back-spacing and rear versus top exhaust conditions. This insensitivity to geometrical changes helped to minimize the number of test conditions. Duplicate thermal performance was also found to exist between the pressurized-panel and suction concepts as shown when comparing figures 13 and 14. These trends of predictable and uniform thermal performance were found to exist over the complete range tested. This is indicated in figure 15 where the maximum heat-flux tested of 1.08 w/in^2 (.167 w/cm^2) is shown. This data illustrates that increased wattage can be compensated for by a corresponding increase in flow rate and at uniform temperatures in the matrix of units.

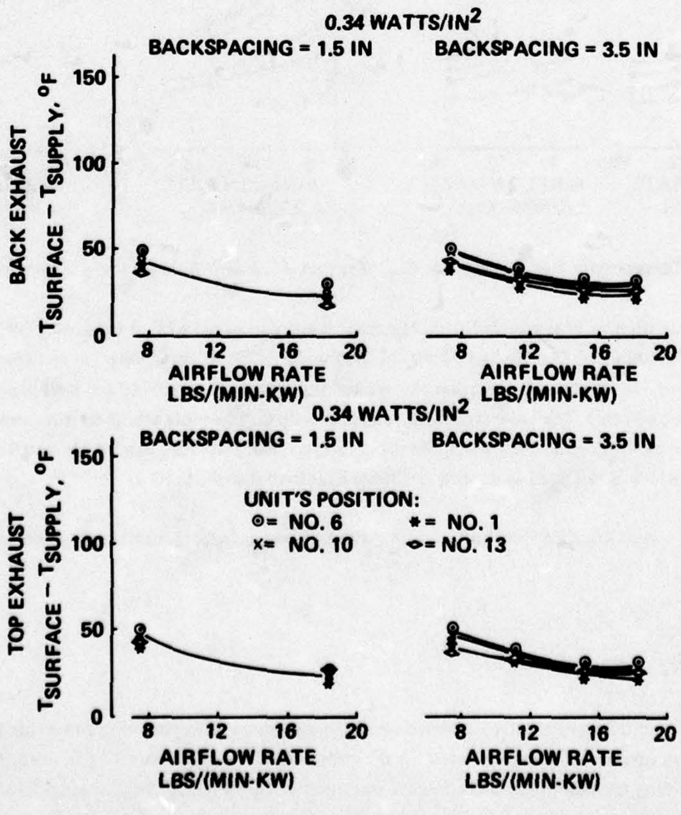


Figure 13.–Temperatures Related to Position, Flow and Geometry–Pressurized Panel Concept–0.34 W/in2

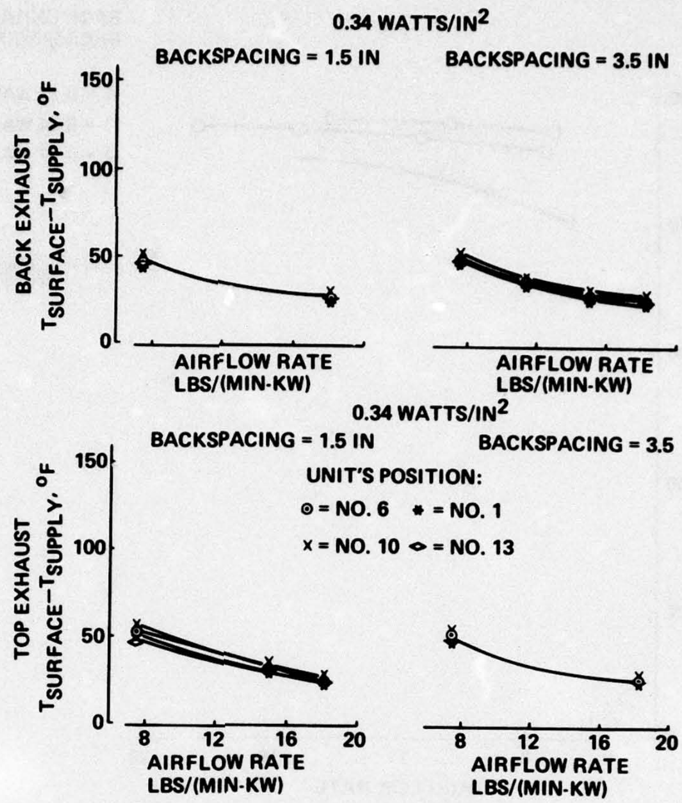


Figure 14.–Temperatures Related to Position, Flow and Geometry–Suction Concept–0.34 W/In₂

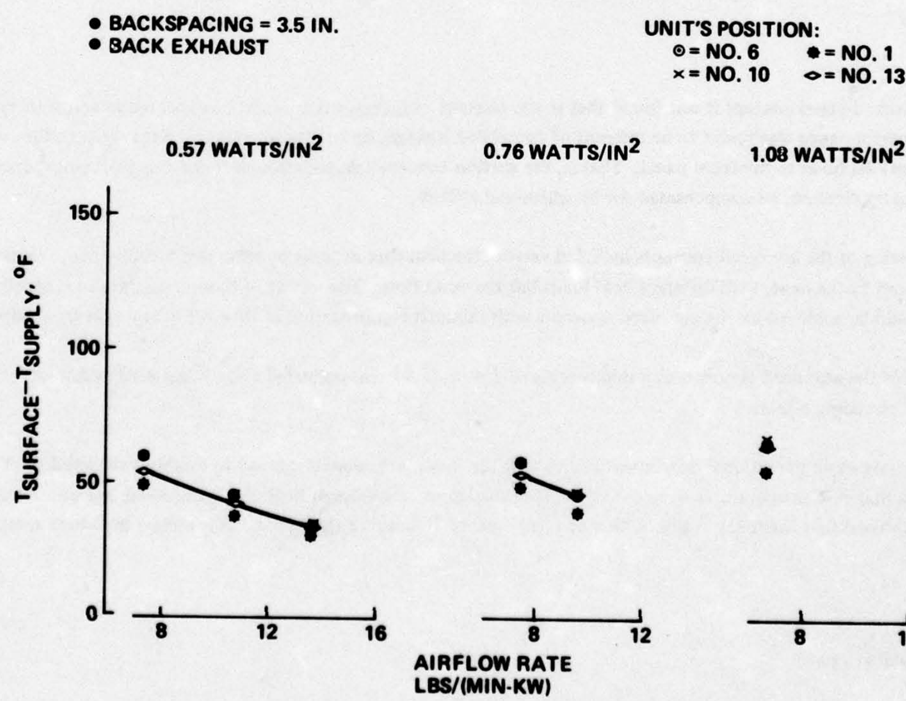


Figure 15.–Thermal Performance at High Heat Fluxes–Pressurized Panel Concept

Figure 16 shows the division of heat rejected from the front panel compared to that exhausted by the cooling system behind the panel. The efficiency of this concept is evidenced by the low percentage of heat rejected back to the laboratory ambient.

Advanced systems were also tested for leakage effects. These tests were conducted to determine the advanced concepts' sensitivity to leakage so that applications of the concepts in actual installations could be better evaluated.

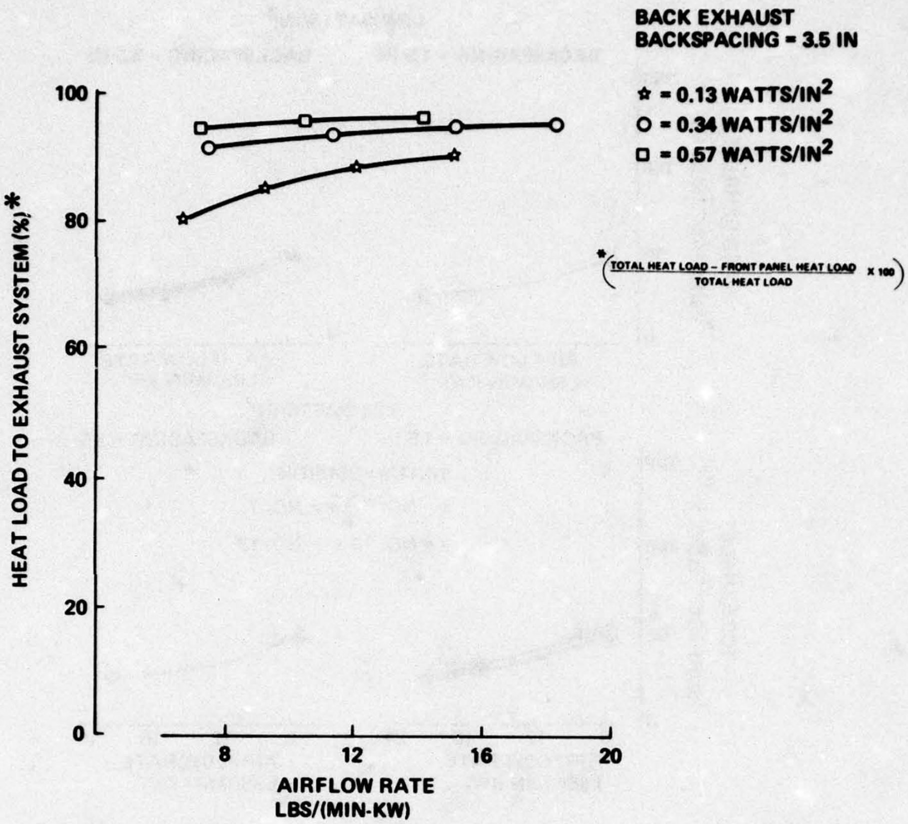


Figure 16.–Heat Load Distribution–Pressurized Panel Concept

For the pressurized-panel concept it was found that it was tolerant of leakages that might be expected in actual installation. The suction concept's performance was found to be tolerant of controlled leakages up to control area openings equal to the total area of the 1/8 in. (.318 cm) dia holes in the front panel. That is, the suction concept's degradation in thermal performance, due to leakage, could, in a practical application, be compensated for by additional airflow.

Additional testing of the advanced concepts included varying the heat-flux of units by rows and by columns to investigate the sensitivity of one unit to the next, with different heat loads but the same flow. The results of these tests showed that independent cooling of units could be achieved by the advanced concepts with minimal compensation in flow for heat-fluxes from adjacent units.

Other testing of the advanced concepts at a side spacing of 3/4 in. (1.91 cm) indicated a slight but predictable increase in temperature and negligible cascading effects.

Additional testing of an operational instrument installed in the test article was conducted to establish the validity of simulation. The results showed that case temperatures were the same along the length and sides of both the operational and simulated unit. In addition, the test showed that the holes in the suction concept's panel, because of their proximity, reduce the touch temperature instrument's bezel.

CORRELATION OF DATA

Figure 17 gives the correlation equation for the pressurized-panel concept. The equation is based on the average heat transfer coefficient (behind the panel) of the 16 units with thermal properties evaluated at the supply temperature. Reynolds number is modified by the ratio of hydraulic diameter to length which was found to provide the best correlation with differnt spacings between units. The hydraulic diameter is also part of Nusselt number and is based on the cross sectional flow area between units. The slope of the curve is indicative of turbulent flow. Evaluation of this data indicates that the holes in the panel act as turbulence promoters which achieve turbulent heat transfer coefficients at low (e.g., $R_e = 410$ or $R_e\ D_H/L = 55$) Reynolds numbers where laminar heat transfer would exist with ordinary duct flow. The correlation with the suction concept is the same, for practical purposes. Similar correlation of the baseline cooling concept was not feasible because of the complex geometrical sensitivity.

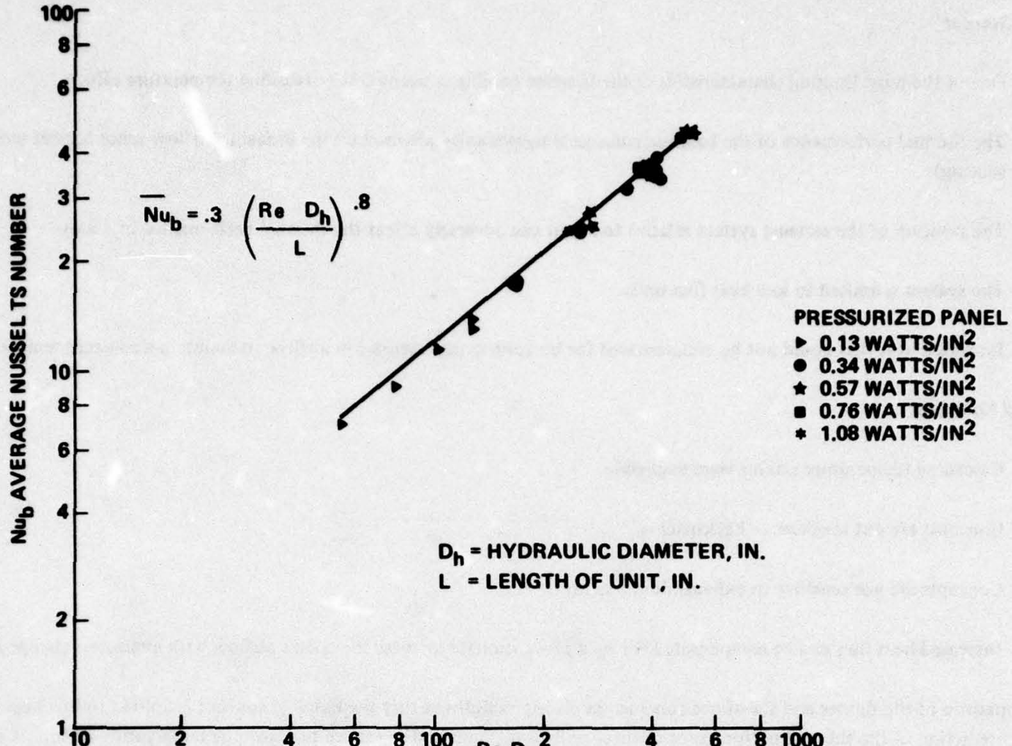


Figure 17.-Correlation Equation for Pressurized Panel Concept

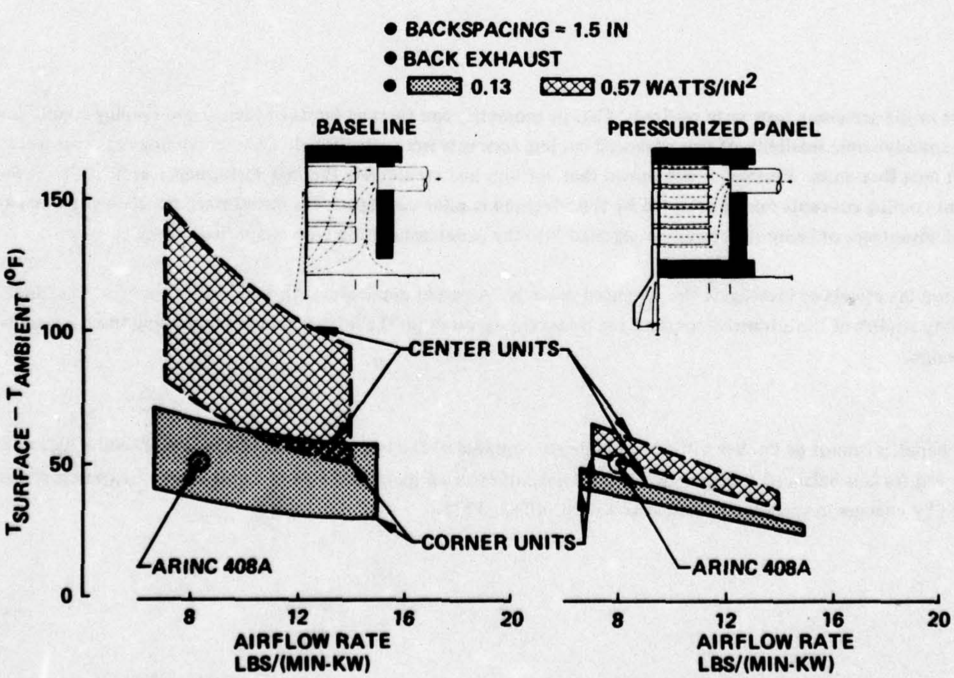


Figure 18.-Comparison of the Baseline to Advanced Cooling Concepts

COMPARISON OF CONCEPTS

The comparison of the baseline concept with the advanced concepts is illustrated in figure 18. From this figure and the detail data presented the following conclusions can be drawn:

Baseline Concept

- One of the most limiting characteristics of the baseline cooling concept is the cascading temperature effects.
- The thermal performance of the baseline concept is significantly affected by the available airflow space behind units (back-spacing).
- The position of the exhaust system relative to a unit can adversely affect the thermal performance of a unit.
- The system is limited to low heat flux units.
- Increased heat flux could not be compensated for by appropriate increase in airflow to maintain a constant temperature rise.

Advanced Concepts

- Cascading temperature effects were negligible
- Concepts are not sensitive to backspacing
- Concepts are not sensitive to exhaust duct position
- Increased heat flux can be compensated for by a proportionate increase in coolant airflow with almost no change in ΔT.

Comparison of the figures and the above conclusions clearly establishes that the baseline concept is limited to low heat flux units. Accurate prediction of the thermal performance of these units is difficult and sensititve to cascading temperature effects. Conversely the advanced concepts showed that uniform, predictable and independent thermal performance of a unit could be achieved at temperature levels acceptable to existing and the proposed specification ARINC 408A, 1975. In addition, this type of performance can be achieved up to the highest heat-flux level tested of 1.08 w/in^2 (.167 w/cm^2) with small increases in temperature.

CONCLUSIONS

The objectives of the screening tests were realized. That is, geometric and thermal limits of the current cooling concepts were defined and the thermodynamic feasiblity of two advanced cooling concepts were established. Current cooling concepts were found to be limited to lower heat flux units. Further, it was shown that uniform and predictable thermal performance at heat fluxes well beyond the limits of current cooling concepts can be achieved by the advanced cooling concepts. The pressurized panel concept, in particular, provides additional advantages of being structurally integrated into the panel installation with simplified clamping.

Boeing is continuing efforts to investigate the advanced systems. A patent application, Groom, 1975, has been filed and mechanical feasibility studies of the advanced concepts are underway to establish the impact that implementing these concepts has on the crew station design.

The true cost and benefits cannot be known without an in-depth appraisal of the internal design of a unit of a similar nature as presented here. The need for this balanced responsibility of internal and external thermal design of a unit cannot be overemphasized and should be reflected by changes in specifications such as ARINC 408A, 1975.

REFERENCES

AEEC letter 75-022/ICM-03, April 4, 1975, "First Draft ARINC Specification 408A," ("Specification No. 408A Air Transport Indicator Cases and Mounting").

Dantowitz, A., May 1971, "Analysis of Aeronautical Equipment Environmental Failures" AFFDL-TR-71-32.

Groom, Kenneth D., December 29, 1975, "Instrument and Panel Cooling Apparatus," Patent Application 645,140.

Military Specification, MIL-E-38453A, 2 December 1971, "Environmental Control, Environmental Protection, and Engine Bleed Air Systems, Aircraft, General Specification for."

DISCUSSION

J Mohr:

Has the use of heat pipes been considered in cooling of instrument panels?

K D Groom and G W Brooks:

For the level of heat flux of current conventional instruments external cooling seems adequate. When we investigated heat pipes analytically it seemed that the installation and cost of heat pipes are one of the main limitations. At present heat pipes tend to be used for specific problems which cannot be met by conventional means.

T V McDonald:

What would be the effect on the cooling system described, of an imperfect fit of the instrument case to the panel due to the typical problem of controlling mechanical tolerances?

K G Groom and G W Brooks:

The absence of a positive seal at the case/panel interface would increase the ΔT by about 10%. Since, however, velocities and delivery pressures are relatively low, adequate leakage control will not be so difficult to achieve. Additionally, the Bezel Assembly could be used to re-inforce the seal.

T V McDonald:

One of the conclusions drawn from your thermal analysis states "Case Temperatures are uniform due to the High Conducavity of the Case". Smaller instruments (2 AT1 sizes) tend to operate in conditions where a high proportion of the total load (eg lighting, motors etc) is dissipated at the front end. Is your conclusion realistic?

K G Groom and G W Brooks:

Before the test, analysis indicated that:-

$$\frac{h\ A\ (\text{inside})}{K\ \frac{A}{X}\ (\text{case})} = \frac{1}{25} \quad \text{and consequently uniform case}$$

Temperatures were anticipated, an operational unit was also tested, and the temperatures recorder at different case locations showed variations in the results (T Surface - T Supply) of less than 10%.

T V McDonald:

In your pressurized panel concept, could you comment on the result achieved relative to the low Reynolds numbers shown?

K G Groom and G W Brooks:

The Re numbers shown were based on the average of 16 units, and the uniform temperature achieved suggested a uniform Reynolds number. The paper (page 11-12) describes the thermal performance and from the evaluation of the results it has been concluded that the unusual flow geometry promotes turbulence and enhances heat transfer co-efficients at much lower Reynolds numbers (400) than is the case with conventional ducting, where the transition range (from laminar to turbulent flow) rarely falls below a Re number of 1200.

THE COOLING OF A POD-MOUNTED AVIONIC SYSTEM

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SUMMARY

The paper describes the principles and testing of the air cooling of the pod-mounted Orpheus day and night aerial reconnaissance system, since 1974 operational with the Royal Netherlands Air Force. The pod airconditioning system was developed by the pod manufacturer Fokker-VFW and flight-tested by the National Aerospace Laboratory NLR.

At moderate airspeeds (e.g. 400 kts), with ram air temperatures up to 44° C, cooling is provided by ambient air entering the pod through a flush air intake in the nose, and leaving it through slots in the pod belly. In low level flight at high subsonic speeds (e.g. 600 kts), with ram air temperatures up to 79° C, the cooling air temperature is kept below 50° C by spraying an automatically controlled amount of water into the airflow near the intake.

During the flight testing of a pre-production reconnaissance system, unexpected cooling problems were encountered. These problems, which had not been experienced during previous prototype tests, could be shown to originate in the NACA flush air intake. By changing to a flush air intake with parallel side walls the cooling system could be made to perform to entire satisfaction in the series.

1. INTRODUCTION

In the first half of 1975 the original aerial reconnaissance equipment of the Lockheed RF-104 G aircraft of a Royal Netherlands Air Force squadron, comprising 3 day-light cameras, was replaced by an advanced pod-mounted day and night reconnaissance system. This so-called Orpheus-system had been designed, in co-operation by the "Old Delft" Optical Industry and Fokker-VFW, to meet air force requirements for a tactical reconnaissance system, providing aerial reconnaissance capability during day and night, at high subsonic speeds and at low to medium altitudes.

The requirement for a pod-mounted system originated from the fact that the space available inside the RF-104 G aircraft was clearly insufficient for the considerable enlargement of the reconnaissance equipment and also from the condition that the aircraft had to remain serviceable and therefore could not be grounded for substantial modifications. A detachable pod provides a solution for these problems and offers the additional advantages of easy maintenance and repair of the pod equipment and, by replacement of the reconnaissance pod by another external store, of mission versatility.

2. ORPHEUS AERIAL RECONNAISSANCE SYSTEM

The Orpheus reconnaissance equipment was developed by the "Old Delft" Optical Industry and the pod structure and aircraft installation by Fokker-VFW. Figure 1 shows the Orpheus pod attached to the standard centre-line bomb rack of a RF-104 G. The only other connection with the aircraft is by way of an electrical connector for power supply lines and remote control by the pilot. This connector has been designed to permit jettison of the pod in case of emergency. Figure 2 gives an impression of the rather densely packed pod interior. It contains from nose to tail:

- a forward looking camera(1) , mounted in a fixed position
- two sideways looking cameras (2 and 4), also in a fixed position
- two more lateral looking cameras (5 and 6) that can be placed in high-oblique or split-vertical positions. In the split-vertical position these cameras, together with the sideways looking cameras, mentioned before, provide complete horizon-to-horizon coverage.
- an infrared linescanner (7) with interchangeable detector package and a total field of view of 120°, roll-stabilized in the vertical and 30° oblique positions
- a control-coupler unit (10) that houses the system interface equipment and the automatic control of camera frame rate, linescanner film speed and airconditioning
- a vertical gyro (12) providing a reference for scanner stabilization
- a static frequency converter (13) that converts the aircraft power supply of variable frequency into fixed frequency power
- a water-tank (14) for the air cooling system
- provisions (3 and 9) for the installation of a radar altimeter, if a true height signal cannot be obtained from the aircraft (not needed for the Netherlands RF-104 G).

In order to benefit fully from the pod configuration, the system should be as self-contained as possible. This implies that the aircraft airconditioning system cannot be used for the pod interior and consequently, in most cases, a separate pod system will be required. Depending on the operational flight envelope, the pod airconditioning system may have to provide cooling at high temperature conditions as experienced in high speed, low level flight, as well as heating during cruising flight at high altitude. Only the Orpheus cooling system will be discussed hereafter.

3. COOLING SYSTEM CONCEPT

According to the environmental specification the Orpheus-system had to be capable of continuous operation at sea level, with sustained true air speeds up to 600 KTAS (knots true air speed) and ambient temperatures up to 30° C. Due to compression and friction of the air flowing past the pod, the temperature of most of its skin rises according to the relation $\Delta T(^{\circ}C) \approx 0.00012\ (KTAS)^2$. From this formula, which could be verified during flight testing, it follows that the temperature rise at the maximum speed of 600 KTAS is 43° C. At an atmospheric temperature of 30° C, this corresponds to a pod skin temperature of 73° C. From these figures and a maximum air temperature of 50° C for cooling the infrared scanner and cameras, as demanded by the manufacturer, it follows that a pod cooling system was called for.

The pod manufacturer Fokker-VFW undertook to develop an airconditioning system providing cooling air

with a maximum initial temperature of 50° C and a minimum mass flow at this temperature of 0.125 kg/sec. The minimum airflow requirement was based upon a total heat dissipation of the reconnaissance system equivalent to 2.500 W and a maximum final air temperature after cooling of 70° C. During extensive ground tests it could be shown that at these temperatures this airflow was just sufficient to warrant proper continuous functioning of the reconnaissance equipment.

With the intention to provide a simple cooling system, needing as little pod volume as possible, Fokker-VFW chose to design an open airflow cooling system, using atmospheric air routed through the pod and kept below 50° C initially by evaporating an automatically controlled amount of water near the air intake. This method of cooling the ventilating air is made possible by the temperature rise due to compression at the intake, which causes the air to achieve a very low relative humidity permitting sufficient water evaporation for the desired cooling effect.

4. COOLING AIR FLOW

Figure 3 shows the general lay-out of the pod air cooling system. Ambient air enters the system through a flush air intake (1), located at the top of the pod nose and subsequently passes through a primary filter (2). This so-called fly-filter is intended to stop insects and coarse particles. It can be easily removed for inspection or replacement through the camera compartment. Downstream of the fly-filter, two water spray nozzles (3) are installed in the air duct. Evaporation of the water takes place in an evaporator unit (4) consisting of two light alloy honey comb parts with a secondary filter in between. This secondary, or main filter collects fine dust and small particles that passed the primary filter (2). These fine particles are largely washed out during flight in rain and during operation of the water spray nozzles. A water separator (5), with drain to the pod skin, is located downstream and partly below the evaporator. It serves to remove rain and the small quantity of water which may not have evaporated at this point of the air cooling process. Beyond the water separator the air passes over a cluster of 4 temperature sensors (6), two of which are used for the cooling process control and the other two for pod heating control and ground measurements. Downstream of the temperature sensors the airflow is divided into 3 directions to provide cooling in the separate compartments of cameras, infrared scanner and frequency converter. After passing the cameras the air is routed backward to the common exhaust through the scanner port, if open, or to the outlet slots in the converter compartment. Distilled water for the cooling system is stored in a 7.6-liter water-tank in the pod tail cone.

5. WATER COOLING SYSTEM

Figure 4 schematically presents the operation of the water cooling system. An electrically-driven pump is fitted to the water-tank. The discharge of this pump is connected to two solenoid operated liquid valves, commanded by temperature sensors in the air duct downstream of the water separator. The solenoid valves pass the water to the spray nozzles upstream of the secondary filter. The pump discharge also has a return to the tank by way of a pressure relief valve, which permits adjustment of the water supply pressure. When the pump is operating, while both solenoid valves are closed, full pump delivery is returned to the tank. The pump circuit is energized when the air temperature reaches the setting of either of the two sensors of the temperature control channels. A time delay relay keeps the pump running for one minute after both temperature control channels have reverted to the de-energized condition, thus preventing unnecessary pump switching.

The primary temperature control channel energizes the pump and opens the primary solenoid valve at a sensor temperature of 44° C, and closes the valve at a temperature which is 1 to 2° C lower. If the air temperature at the intake is only slightly above 44° C the water cooling will function intermittently. As temperature and airflow increase, for instance due to increasing flight Mach number, the spraying of water will continue for a greater percentage of time till the point where the water evaporation cannot any longer lower the temperature below the de-activation temperature of approximately 42° C. The primary nozzle then is spraying continuously. The secondary temperature control channel opens the secondary solenoid valve when the temperature at the corresponding sensor rises to 48° C, and closes the valve at a sensor temperature which is 1 to 2° C lower. Operation is similar to that of the primary channel. During intermittent operation of the secondary channel, the primary channel remains continuously activated.

6. PROTOTYPE TESTING

For the development of the cooling system, a closely interrelated series of ground and flight tests had to be carried out. The flight testing started with a dummy pod, without reconnaissance equipment, to locate the best position and dimensions of the air intake for meeting the minimum cooling airflow requirements. The dummy pod contained an air-duct with resistances to simulate the cooling unit and reconnaissance equipment and was instrumented to measure airflow as a function of flight Mach number. The range of measured airflows, corresponding with the range of airspeeds for reconnaissance missions, was then used in test rig experiments for the detail design of the water cooling system.

The test rig consisted of a complete cooling unit assembled in a representative duct with flush air intake and provisions for pressure and temperature measurements at several points in the airflow. Aft of the cooling unit, the underside of the duct was constructed of a transparent material to enable detection of possible free water in the airstream. A temperature controlled air supply to the flush intake of the duct was provided by a Roots blower and electric heating elements. The airflow was measured with a calibrated nozzle.

During the test rig experiments it proved to be rather difficult to find the right solution for the handling of the large range of possible airflows (0.10 to 0.18 kg/sec) and temperatures (40-72° C). For instance, with a nozzle configuration providing adequate cooling capacity at maximum flow and temperature, sometimes small amounts of free water were detected at minimum flow and at switch-over from operation with one to two nozzles. Eventually, however, the correct combination of waterpump pressure, nozzle size and switching temperatures was attained and employed in the cooling system of the Orpheus prototype. In order to achieve the same cooling performance in flight as on the ground, the airflow in the prototype had to be adjusted to equal the rig test values. During the first few flights with the complete reconnaissance equipment installed and operating, this was accomplished by adaptation of the resistance of a grating (perforated steel plate) fitted for that purpose in the air intake.

After extensive flight testing in the Netherlands and some small adjustments to the cooling system,

final testing took place in central Italy, where high ambient temperatures allowed flight tests up to the limits as stated in the environmental specification. Final proof of the maximum cooling capacity was obtained during a flight at 600 KTAS continuous, a height of about 500 ft and an ambient temperature of nearly 30° C. During this flight, which lasted 50 minutes, all temperatures in the pod remained within limits, and the reconnaissance equipment functioned satisfactorily.

7. AIR INTAKE PROBLEMS

In summer 1973 a pre-production Orpheus system, with modified camera configuration but essentially the same pod structure and airconditioning as the prototype, became available for flight testing. During the first few flights the water cooling system turned out not to be working properly. Although repeated ground tests did not show any deficiencies, and the water consumption in flight was exactly as it should be, the cooling effect, if any, was negligible and during short runs at 600 KTAS, air temperatures up to 70° C were recorded downstream of the evaporator. When, after having checked everything without finding anything wrong, in a last effort the air intake of the pre-production pod was replaced by the one of the prototype, the water cooling resumed normal operation! A careful comparison of the air intakes only showed a slightly different location of the grating used for airflow trimming. Evidently small alterations to the intake configuration could be of great influence on the evaporation process.

As it was the intention for the production series to dispense with the intake grating and to introduce instead a readily replaceable fly-filter further downstream, it was decided to interrupt the flight testing in order to first introduce this comparatively large modification.

Flight testing with the proposed series intake configuration was resumed in autumn 1973. The water cooling thereby proved to perform normally without, however, attaining the same low temperatures as during prototype measurements. This soon turned out to be caused by substantially higher air intake temperatures than could be accounted for by flying speed and atmospheric temperature.

If all kinetic energy of flowing air is transformed into heat by reducing its speed to zero, the temperature rise due to adiabatic compression amounts to $\Delta T(^\circ C) = 0.000132\ (KTAS)^2$. The fraction of this maximum temperature rise, that actually occurs at a certain location is called the local recovery factor. During the prototype flight testing the recovery factor r of the air, downstream of the intake, just before reaching the spraying nozzles, was measured several times and consistently proved to be r=0.85. Now, however, with the series intake configuration, using a fly-filter instead of a grating, a recovery factor r=1.1 was found! At the maximum speed of 600 KTAS the increase of recovery factor meant that the water cooling system was called upon to handle a 12° C higher temperature than expected and designed for.

Since a recovery factor above 1.0 at first seemed to be highly improbable, quite a number of flight tests were made, even using independent measuring systems, which, however, confirmed the first results. The explanation for this unusual phenomenon most probably is to be found in the complicated flow pattern inside the so-called NACA air intake as chosen for the Orpheus prototype. This type of air intake, shown in figure 5, was selected because of its ability to produce relatively high intake pressures, and its immunity from icing up. The efficiency of the NACA air intake is based on the generation of rather strong vortices, which, in certain circumstances, are known to be capable of producing abnormal temperature distributions (Ranque - Hilsch effect). Also, an unsteadiness in the airflow of the resonance type could be responsible for the temperature build-up.

In order not to be forced to redesign the water cooling system, the testing of the Orpheus pre-production system was concentrated upon the retrieval of a low recovery factor. Several intake configurations, with and without different types of grating and filter, were tested in flight as well as at full scale in a high speed wind tunnel of the NLR. Within the limits set by time and budget, however, a recovery factor r=0.85 could not be regained consistently.

As both the total loss of cooling at the start of the flight testing, and the excessive recovery factor experienced later on, were clearly connected with the NACA air intake, it was finally decided to change over to a simple straight air intake as shown in figure 6. This type of air intake with parallel side walls generates no vortices and is known to produce a steady recovery factor r=1.0 under all conditions, which was confirmed during Orpheus flight and wind tunnel tests. As a consequence the cooling capacity had to be increased to cope with the higher recovery factor, and the filter resistance had to be decreased to keep up the minimum cooling airflow. Eventually, however, a satisfactory configuration could be attained.

8. SERIES PERFORMANCE

In the final series configuration the limiting condition for the cooling system occurs at sea level with a sustained flight Mach number M=0.90 (610 KTAS) and an ambient temperature of 30° C. The total cooling airflow under these conditions is 0.167 kg/sec with an intake temperature of 79° C. Both spray nozzles are operating continuously, with a total water consumption of approximately 2.5 cm^3/sec, permitting 50 minutes of flight under these conditions at the given tank capacity of 7.6 liters. By evaporating 2.5 cm^3 of water in 0.167 kg of air the humidity is increased by 15 g/kg. Consequently, even while flying in a saturated atmosphere at 30° C (containing 27 g/kg) the humidity downstream of the cooling unit will not exceed 42 g/kg, that is 49 % relative humidity at 50° C. Since in theory the cooling of 0.167 kg of air from 79° C down to 50° C could be achieved by complete evaporation of 2.1 cm^3 of water, the cooling process can be said to have an efficiency, under these conditions, of 84 %.

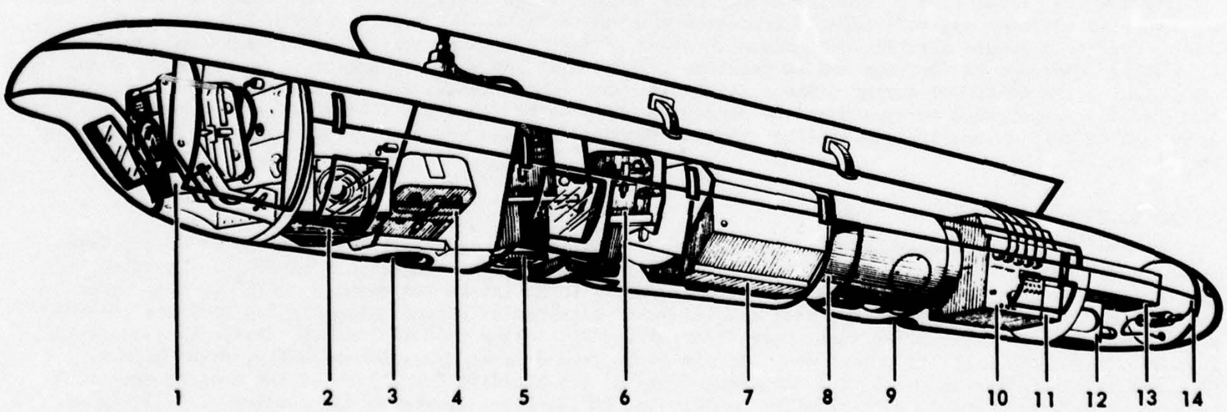
At a flight Mach number M=0.60 (405 KTAS) at sea level and an ambient temperature of 30° C, the cooling airflow with the straight air intake was measured to be 0.094 kg/sec with an intake temperature of 52° C. Intermittent operation of the primary nozzle reduces this temperature to 44° C, thus providing just sufficient cooling capacity.

9. CONCLUDING REMARKS

For one and a half years now, the Orpheus reconnaissance system has been fully operational. During a considerable number of sorties, in all kinds of weather, the airconditioning system worked as it was required to do. From a practical point of view, the final configuration therefore can be said to be satisfactory. From the experiences during the flight testing of the Orpheus prototypes, it can be learned, however, that the temperature behaviour of cooling air intakes should be checked carefully, since small alterations of the flush NACA intake turned out to be capable of producing rather surprising results. A closely-

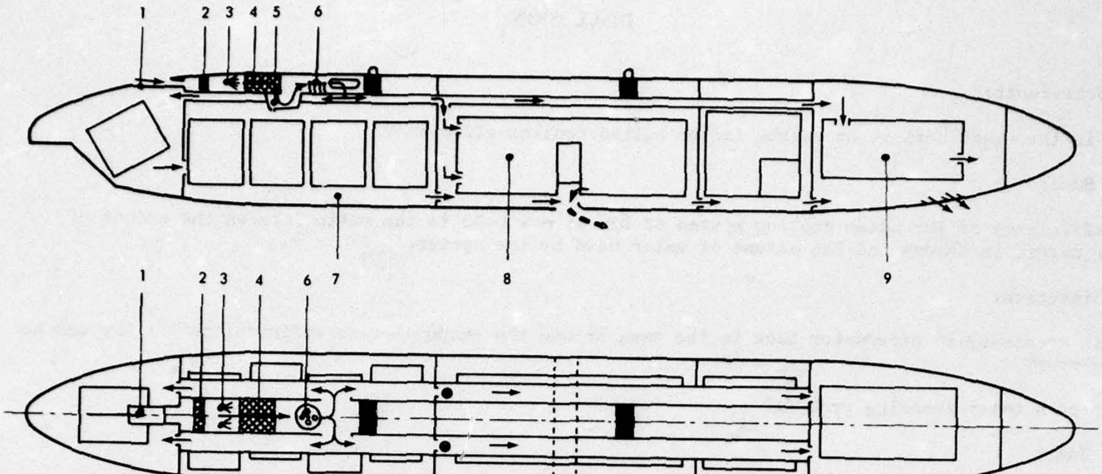
reasoned explanation for this behaviour is still missing. Therefore, the inquisitive mind of the research-worker is not satisfied. He still hopes, of course, to be given the opportunity some time to solve this problem.

FIG. 1 ORPHEUS RECONNAISSANCE POD ATTACHED TO A LOCKHEED RF-104G



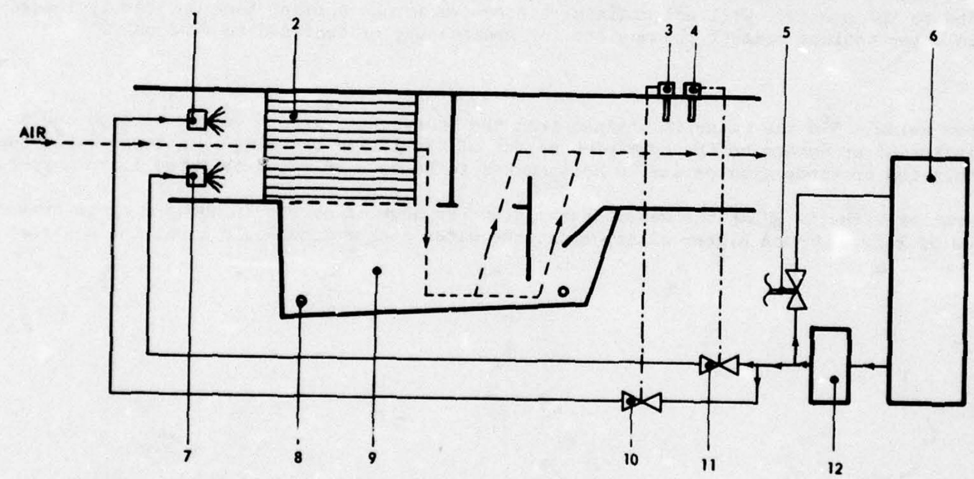
1. FORWARD CAMERA
2. LEFT PRIMARY CAMERA
3. RADAR ALTIMETER Tx ANTENNA
4. RIGHT PRIMARY CAMERA
5. LEFT SECONDARY CAMERA
6. RIGHT SECONDARY CAMERA
7. INFRARED LINESCANNER
8. SCANNING PORT DOOR
9. RADAR ALTIMETER Rx ANTENNA
10. CONTROL-COUPLER UNIT
11. TEMPERATURE CONTROL BOX
12. ROLL REF. GYRO
13. AC/AC CONVERTER
14. WATERTANK AND PUMP

FIG. 2 LAY-OUT OF THE ORPHEUS RECONNAISSANCE SYSTEM



1. FLUSH AIR INTAKE
2. PRIMARY FILTER
3. WATER SPRAY NOZZLES
4. EVAPORATOR WITH SEC. FILTER
5. WATER SEPARATOR
6. TEMPERATURE SENSORS
7. CAMERA COMPARTMENT
8. INFRARED SCANNER
9. FREQUENCY CONVERTER

FIG. 3 COOLING AIR FLOW IN THE ORPHEUS RECONNAISSANCE POD



1. SECONDARY SPRAY NOZZLE
2. EVAPORATOR AND SEC. FILTER
3. TEMPERATURE SENSOR 48° C
4. TEMPERATURE SENSOR 44° C
5. PRESSURE RELIEVE VALVE
6. WATER-TANK
7. PRIMARY SPRAY NOZZLE
8. DRAIN TO POD SKIN
9. WATER SEPARATOR
10. SEC. SOLENOID VALVE
11. PRIMARY SOLENOID VALVE
12. ELECTRIC WATER PUMP

FIG. 4 WATER COOLING SYSTEM

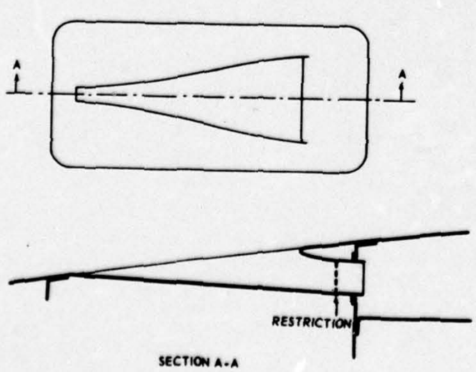


FIG. 5 NACA AIR INTAKE FOR PROTOTYPES

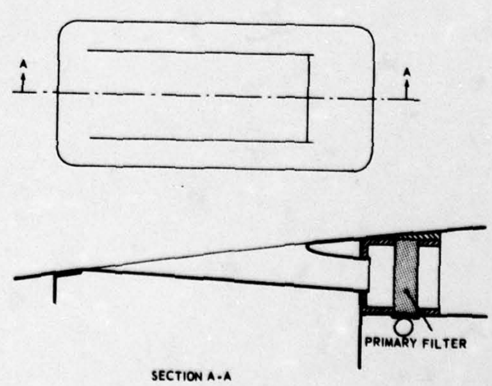


FIG. 6 STRAIGHT AIR INTAKE FOR SERIES

DISCUSSION

W J Schwarzott:

What is the exact definition of the factor called cooling efficiency?

I de Boer:

The efficiency of the water cooling system of 83% at M = 0.90 is the ratio between the amount of water needed in theory and the amount of water used by the system.

G F Stevenson:

Was it necessary to pipe water back to the tank or was the evaporator so efficient that there was no carry over?

Is there a water freezing problem?

I de Boer:

In the prototype excess water was indeed piped back to the tank. In the series this pipeline was omitted, as the total water consumption per flight of one hour (in actual operational use) turned out to be only about 2 kg (with a water tank capacity of 7.61).

There is no freezing problem, since water cooling is no longer needed as soon as atmospheric temperature drops below + 5°C (for speeds up to 600 KTAS), when the water tank is drained. In Holland this has to be done 2 or 3 times per year, which was not considered unacceptable by the RNLAF.

F S Stringer:

Following Mr Stevenson's question, will not freezing occur in winter at dispersal points? Can anti-freeze be added to the system? Will not draining the system impose another task on heavily loaded ground crew? Is this water-coolant concept recommended for general use or confined to PODS only?

I de Boer:

At temperatures below + 5°C the water is drained from the tank. Anti-freeze cannot be used due to possible pollution of or damage to the germanium optics of the IRLS. Draining is a very simple task which takes only two or three minutes and in Holland has to be done only two or three times a year.

In actual operation with the RNLAF the water consumption per hour of sortie is about 2 kg in summer for a recce system of 2 kW. For much higher dissipations the water consumption would limit the system.

EFFICIENT SOURCES OF COOLING FOR AVIONICS

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SUMMARY

Conventional systems for providing cooling for airborne avionics tend to use cooling methods similar to those used in cabin airconditioning systems, and indeed, in many aircraft the cabin conditioning system provides a portion of its air supply directly for avionic cooling.

Such a cooling system is almost invariably of the open air cycle type using air bleed from the main propulsion engine. These systems are chosen on the basis of low weight and volume, together with the availability of conventional equipment rather than low power consumption and their design is also constrained by the necessity to provide fresh air to ventilate the cabin.

Much work is being done to reduce the bleed airflow requirements of cabin conditioning systems, but it is often possible to make even more significant savings if the design is aimed at avionics cooling at the outset.

This paper discusses the penalties and design constraints of the present day approach and describes some new systems which can be applied with advantage to avionics cooling.

These range from orthodox air cycle bleed air systems to those utilising ram air only for their power supply.

The emphasis is on systems designed specifically for avionics cooling with minimum overall aircraft penalty.

The effect of the aircraft operational role is also discussed.

1. INTRODUCTION

As the avionic load of modern aircraft increases, so does the power required to produce the necessary cooling. Even simple systems using direct ram air have to pay a penalty in aircraft drag whilst those using air bled from high performance engines can cost far more in power than the power used by the avionics themselves. As an example, the total power penalty for an air bleed system cooling an avionic load of 10kW on a Mach 2.0 aircraft could easily be 200kW. Thus there is every incentive to make the cooling system as efficient in power consumption as possible.

2. PRESENT DAY COOLING SYSTEMS

There is a wide range of methods of cooling avionics at present in use, and this is to be expected when the differences in aircraft role, speed range, size and so on, are considered.

In many cases the avionics are cooled by ambient air as shown in Fig.1A and for aircraft with low forward speeds this can be acceptable. The helicopter is a typical example of this where, to date, no special systems for cooling are used. Even for helicopters there is a growing awareness that increased avionic heat dissipation and higher packaging densities will lead to reconsideration of the situation and the provision of special cooling systems.

The extension of this method is to place the avionics packages in the aircraft cockpit or cabin (Fig.1) where they receive the benefit of the cooling already provided for the people in those enclosures. As the avionic heat load grows relative to the remaining cabin load, then the system must be increased in capacity to provide the extra cooling. The temperature requirements of electronic equipment are rarely identical to those of the human body and so a cooling system designed to provide personal comfort is not necessarily ideal for heat removal from avionics. In fact avionics will normally operate satisfactorily at higher temperatures than required for comfort and so it is not economical to keep them cooler than necessary.

By cooling the electronics in series with the cabin (Fig. 1c) the above objections to placing the avionics in the cabin can be overcome. The air leaving the cabin is cool enough to do useful cooling in the avionics before being passed overboard. There should be no recirculation of the air from the avionics back to the cabin. Provided that the avionics air demands are no more than those of the cabin alone then this is a very economical cooling system and can be regarded as a zero power penalty solution. The main difficulties are those of installation and the possible adverse effects on the cabin pressure control system.

Where for various reasons the above systems cannot be used (e.g. difficult location of avionics bays) or are insufficient, then it is common practice to use an oversized air system for the cabin and duct some of this air to the avionics directly (see Fig 1d). With this arrangement the full cooling potential of the conditioned air is available to the avionics, but the demands of the cabin temperature control system may mean that heating is supplied to the cabin when cooling is required by the avionics. The design of

of such a cooling system would have to be able to allow for this and so could be wasteful on cooling in some cases.

The logical way of improving this situation is to divorce the cabin and avionic cooling systems and many high performance aircraft flying today do have cooling systems dedicated to avionics cooling alone.

The rest of this paper is devoted to this approach and includes a discussion of the advantages of separate avionics cooling systems.

3. AVIONICS COOLING REQUIREMENTS

At this point it is worth reviewing the fundamental requirements of an avionics cooling system and also noting where these requirements differ from those of cockpit and cabin conditioning systems. This should enable full advantage to be taken of the peculiarities of avionic systems and so provide the cooling with a minimum penalty on aircraft performance.

It has to be appreciated of course that there can be no single definition of requirements as avionic systems themselves differ considerably in their cooling requirements.

It follows also that there can be no one solution to the general cooling problem which provides a minimum penalty system.

The question of heat dissipation within the avionics will be discussed in other papers and is generally outside the scope of this presentation. For example, in many cases cooling air is required to be supplied direct to the equipment, whereas in other situations particularly in modern, densely packed arrangements, an intermediate heat transfer fluid is used to transport heat from the avionics to a suitable heat sink.

It is the purpose of this paper to describe means of providing such a suitable heat sink.

The main requirements of an avionics cooling system are given below:-

Cooling only is normally required (Heating may occasionally be necessary for pre-heating and drying)

Temperaturelevels may be 80°C or so

The cooling system is not normally required to provide pressurisation

Free water must not enter the avionics

Indirect cooling can often be used

Close limit temperature control is not normally required

Airflow rates are determined by heat transfer

The main differences between these requirements and those of cabin airconditioning are that:-

Cabin heating is required over the majority of the flight envelope (except for high speed aircraft

Temperature levels in the cabin are more normally required to be 20-25°C

Pressurisation is determined by physiological conditions and an absolute pressure of 0.35 bars is about the minimum

Limited free water can be allowed to enter the cabin

Indirect cooling is hardly ever used, as the fresh air provides ventilation

Temperature control to within 1°C of the set point value is usually required

4. PENALTY EVALUATION

In the comparison of various cooling systems it is ideal to reduce the various penalty parameters to a common basis. The most satisfactory one is probably Take-off Gross Weight. The power loss caused by bleeding air from an engine and the use of shaft power can be equated to fuel flow. The extra power required from the engine to overcome aircraft drag caused by ram air usage can also be translated into fuel flow. The effects of equipment deadweight again can be related back to extra fuel required to produce the lift by the assumption of the lift/drag ratio, and the whole of the extra fuel usage can be integrated over a particular aircraft flight mission (including fuel required early in the mission to carry fuel used later) to give a total weight of fuel required at take off time. The dead weight of the equipment is again added to give the total Take off Gross Weight.

To do this type of assessment requires a fairly good knowledge of the aircraft role and the engine/airframe relationship, i.e. lift/drag ratios', specific fuel consumption etc. To simplify things for this paper the assessment has been done only on the actual power penalty at a given instant of time. The power used in taking ram air on board the aircraft and the power used to provide bleed air are shown on

Figs. 2, 3 and 4. The power losses used in comparison of systems later in this paper were taken from these figures.

For comparison purposes all systems discussed in this paper are based on an avionics load of 10kW and maximum coolant temperatures of 50°C and 80°C (This is the temperature of the coolant leaving the avionics). Where an intermediate coolant liquid has been used the temperature is the temperature of the air leaving the heat exchanger.

Sea level operation on a 40°C day and 40,000ft operation with an ambient temperature of -40°C are both considered.

5. COOLING SYSTEMS CONSIDERED

The rest of this paper is devoted to comparing and commenting on various candidate cooling systems which will provide a low temperature heat sink for the heat dissipated by avionic systems. As discussed earlier, only dedicated avionic cooling systems are considered.

The candidate systems chosen are:-

1. Ram Air
2. Fuel Heat Sink
3. Turbo-fan System
4. Bootstrap System
5. Reversed Bootstrap
6. Ram Powered Reversed Bootstrap

An attempt is made to present some values of the extra power required to be provided by the powerplant (whether in compressing bleed air or providing thrust to overcome ram drag) so that the usefulness of each system for a given application can be assessed.

In all cases the powers quoted are the minima, i.e. the system is assumed to be designed for the particular bleed pressure,speed and altitude quoted. Although this is of limited value in itself, it does allow useful comparison of systems.

5.1. Ram Air Cooling

Although ram air is not 'free' in the sense that drag is caused in taking ram air into the aircraft, nevertheless it requires only the simplest of associated systems and should always be considered as a candidate cooling medium. Whether or not it can be used depends primarily on the temperature of the air when brought to rest relative to the aircraft. Fig.5 shows the temperature of ram air as a function of Mach number. This shows that, whereas ram air has limited use at sea level, it can provide a very useful cooling function at the higher altitudes. The cost in terms of aircraft power in using ram air is shown on Fig. 6.

5.2. Fuel Heat Sink

The use of fuel as a heat sink for the avionics can be quite attractive, especially when compared with using fuel as a heat sink for cabin conditioning. In the latter case fuel cannot be used directly because its temperature level is often above that required for the cabin, whereas for avionics, particularly at the higher temperature levels the heat can pass from the avionics to the fuel with no intermediate cooling system (except perhaps a heat transfer fluid),

Whether or not fuel can be used will depend very much on the fuel tank capacity, heat input to (or loss from)the fuel from other systems and aerodynamics causes and the flow of fuel to the aircraft engines.

There may well be a case for a fuel cooled avionics cooling system which incorporates an additional refrigeration system to remove the excesses of temperature in the fuel. Because this latter system would only need to operate occasionaly, its total penalty on the aircraft could be relatively low. This type of approach has been used on the Rockwell B1 aircraft (STEIN and SCHEELE, 1975).

Because of the above variables, no specific penalties for fuel cooled systems are included in this report but it is certainly a method of cooling which should be given full consideration.

5.3. Turbo-fan System

This is the simplest form of air cycle cooling system and is often referred to as a 'simple Cycle' system. It is shown diagrammatically on Fig.7.

In this system air bled from a high pressure source (usually the compressor of the main propulsion engine) is cooled by ram air in an air to air heat exchanger. The air is then expanded through a turbine where its temperature is further reduced, thus producing a flow of air at a temperature well below that of the ram air. The power produced in the turbine is used to drive a fan mounted on the turbine shaft and this fan assists the flow of ram air through the heat exchanger.

In any air cooling system the heat absorbed by the cold air supply is the product of the air mass

flow rate, the air temperature rise in passing through the load, and the specific heat of the air. The maximum allowable temperature of the air leaving the load can be regarded as fixed by the temperature limitations of the electronic components, and so variations in temperature rise mean variations in required inlet temperature. Thus a given amount of cooling can be provided by a low airflow at low temperature or a higher flow at higher temperatures.

As the temperature of the air delivered by the turbo-fan system depends mainly on the input pressure and the ram pressure, it is possible to determine the flow required to produce whatever cooling is required, and the power penalty can then be estimated.

Fig.8 shows how the estimated power required to produce 10kW of cooling would vary with Mach Number and bleed pressure for the sea level and 40,000ft cases.

In each case it can be seen that at the low Mach Numbers (low ram air temperatures) there is an optimum value of bleed pressure at a relatively low value, but as the Mach Number increases so must the bleed pressure if the penalty is to be minimised.

By plotting similar figures for a 50°C maximum avionic temperature as well as the 80°C value used in Fig.8, it was then possible to produce Fig.9 which shows how the minimum power to drive the turbo-fan cooling system varies with Mach Number at both temperature levels.

5.4. Bootstrap System

In forward flight, the main purpose of the fan in a turbo-fan system is to absorb the turbine power, there usually being sufficient ram pressure to induce cooling flow through the turbine. The bootstrap system shown on Fig.10 uses the power from the turbine to drive a compressor which boosts the pressure from the bleed air before passing it through the heat exchanger to turbine inlet.

Because of mechanical limitations it is normally necessary to pre-cool the bleed air before it enters the compressor of the bootstrap unit and this increases the ram air penalty of such a system.

Fig.11 shows the minimum power required for such a system on the same basis as that shown in Fig.9 for the turbo-fan system.

5.5. Reversed Bootstrap System (Bleed)

The use of ram air is very expensive on power at the higher speeds and so systems which minimise the amount of ram air required must be given due consideration. One such system is the reversed bootstrap system shown on Fig.12.

As its name implies it uses the turbine/compressor arrangement of the normal bootstrap system, but the bleed flow passes through the turbine before the compressor in this case. The compressor serves to depress the pressure at turbine outlet and allow the use of a regenerative heat exchanger in the line. It is unlikely that such a system could completely dispense with ram air cooling except perhaps at low bleed pressures (hence temperatures) and in this case a small ram air unit has been assumed.

The power required to operate this type of system is plotted as Fig.13 for the two levels of avionics temperatures used in this paper.

This performance will be compared with other systems later in the paper.

5.6 Ram Powered Reversed Bootstrap

It can be seen from Fig.6 that, where ram air can be used for cooling the avionic load, the power penalties are not very high. If the ram air is collected in an efficient pressure recovery intake then it is possible to extend the useful range of ram cooling by expanding this ram air through a cooling turbine (Fig.14).

The potential of this system is demonstrated in Fig.15 which shows how much power would be required to operate such an expanded ram air cooling system. Comparison with Fig.6 shows that the system can provide cooling over a much wider range of speeds than the pure ram system.

In practice it would be difficult to design a system which could provide the cooling at the higher forward speeds because the airflows required become large, calling for a very large cooling turbine. However, if the power from the turbine is used to drive a compressor in a reversed bootstrap arrangement, then the extra pressure ratio across the turbine enables a lower airflow to be used. The layout of the ram powered reversed bootstrap system is shown in Fig.16.

Figure 17 shows the avionic temperature variation and the power penalty of a practical realisation of the ram powered reversed bootstrap system. The cause of the reduced performance at high Mach Numbers is choking of the compressor. A variable nozzle turbine would improve this situation.

6. COMPARISON OF SYSTEMS

A comparison of the three bleed air systems with the power penalties of ram air systems are shown on Figure 18. The penalties are compared for the case of 80°C avioncs temperature, although the trends will be similar for the 50°C temperature.

7. SPECIAL FEATURES

Where bleed air systems are used then it is most efficient to use the highest turbine expansion ratio possible. This could easily lead to sub-zero temperatures at the turbine outlet, with the possibility of ice formation downstream. This can be efficiently overcome by mixing with warm air from the avionics outlet as shown on Fig.19. Experimental work on effective mixing arrangements is underway at this time.

The bleed air systems can be used to provide cooling on the ground but the ram air systems require forward speed to operate. This latter problem can be overcome by providing a supply of low pressure air or, alternatively, putting shaft power into the ram powered bootstrap. This then becomes a powered bootstrap on the ground and development of a suitable system is underway at this time.

8. CONCLUSIONS

The main object of this paper is to give a comparison of the power penalty of a number of approaches to the problem of providing cooling for avionics for advanced aircraft.

Many questions have been left unanswered, particularly those concerned with the practicality of designing such systems to operate over the many off design cases. However it is hoped that the work presented in this paper will help to point the way to reducing the power penalty of cooling avionics.

One conclusion easily seen is that ram air is a very expensive user of power and it is very uneconomical to use it as a heat sink for cooling bleed air. Perhaps surface heat exchangers (LE CLAIRE, 1976) could help here.

Systems which use cooling already available for other purposes, such as cabin exhaust air and fuel cooling should always be considered for use. Regeneration of the air leaving the avionics or the load heat exchanger will show savings, especially on high speed aircraft.

Using maximum allowable avionics temperatures will again reduce the power penalty, although the effects on avionics life must be considered.

Efficient use of the cooling is essential. This involves such things as series cooling of certain components so that the most critical parts are kept at the correct temperatures.

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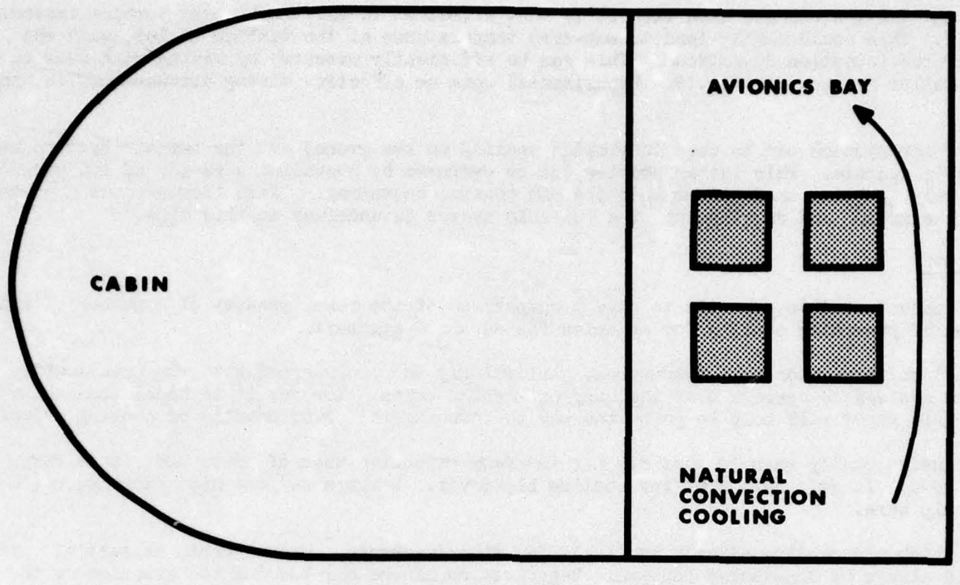
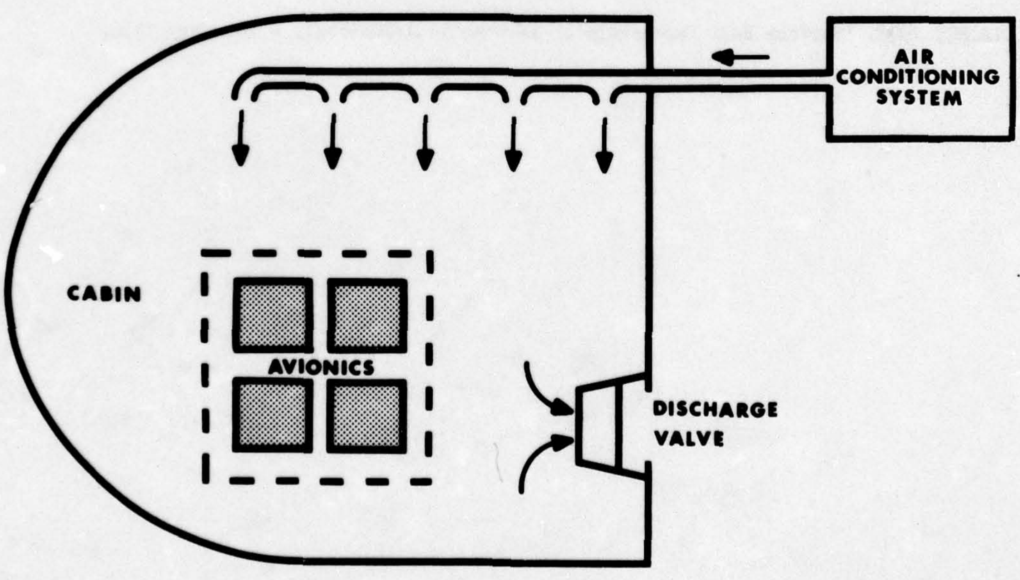
STEIN.P.G. and SCHEELE.L.,1975 'The B1 Environmental Control System' - ASME 75-ENAs - 13.

LE CLAIRE, 1975, 'Surface Heat Exchangers'. Aeronautical Quarterly - February 1976.

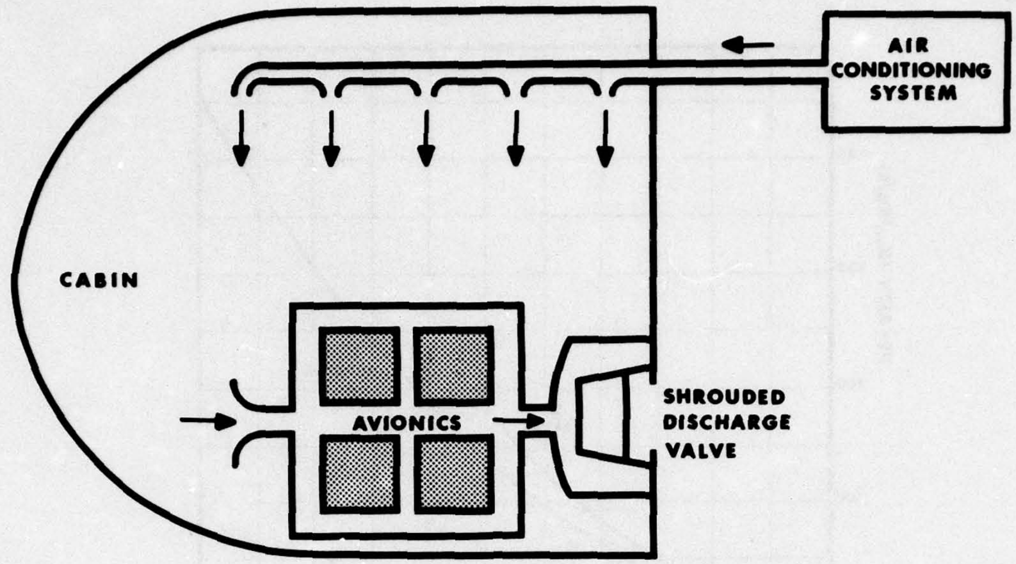


(a)

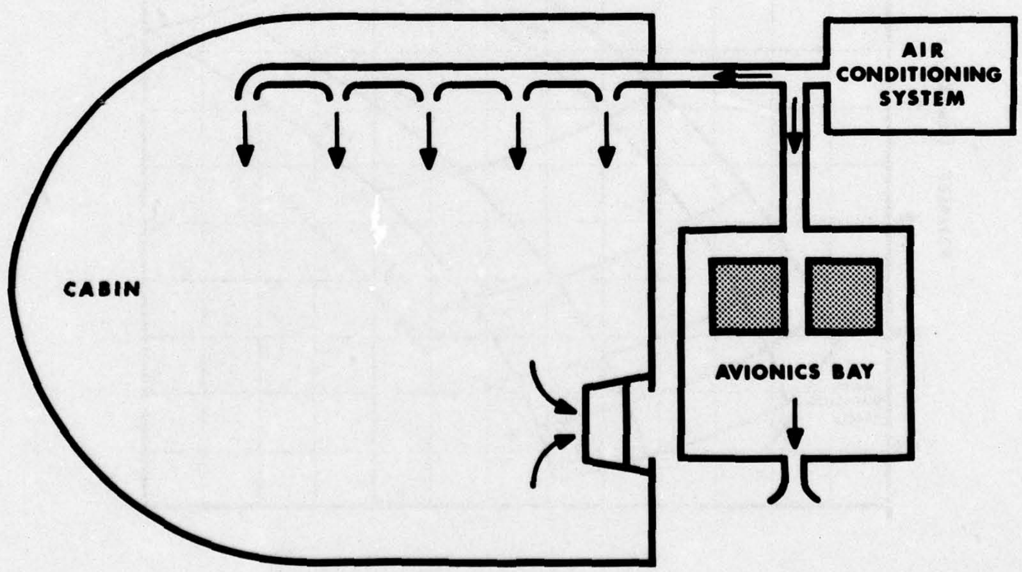


(b)

FIG. 1 SOME EXISTING AVIONICS COOLING SYSTEMS



(c)



(d)

FIG.1 SOME EXISTING AVIONICS COOLING SYSTEMS

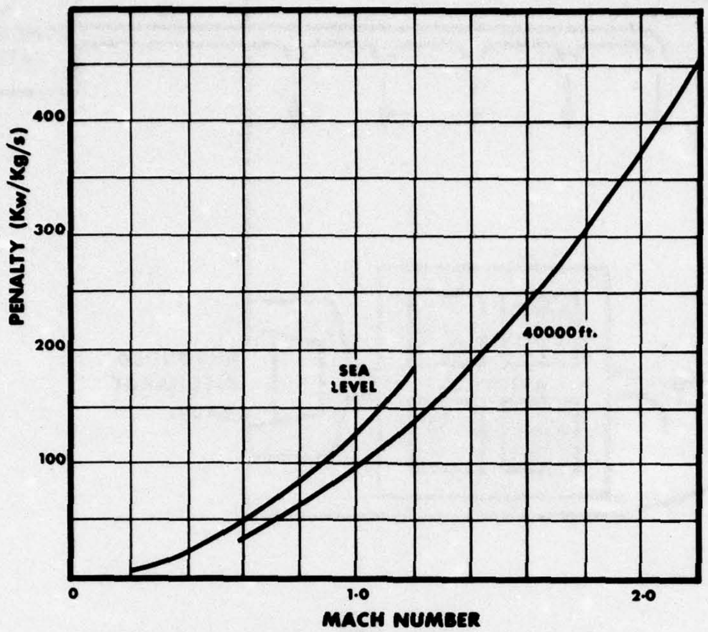


FIG. 2 PENALTY OF USING RAM AIR

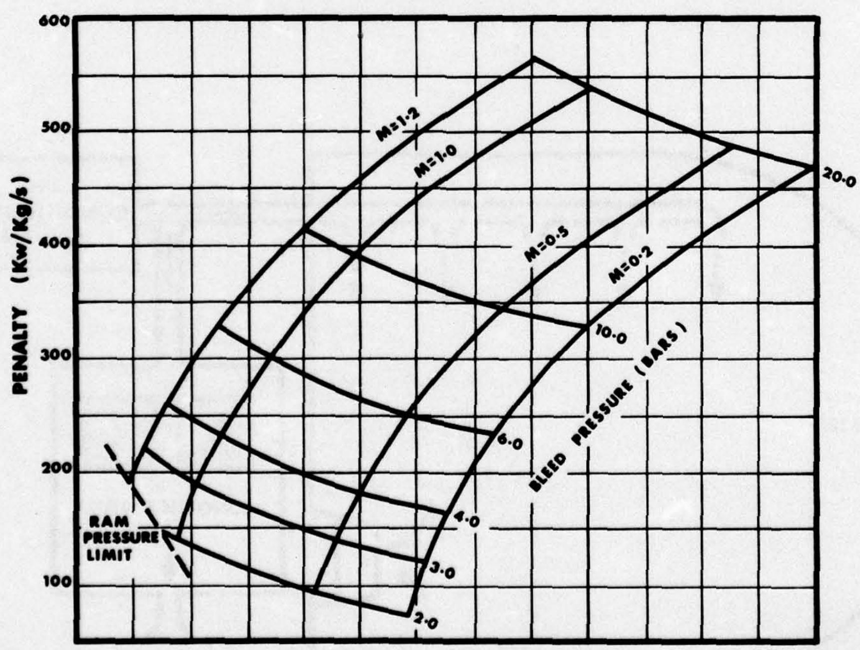


FIG. 3 PENALTY OF USING BLEED AIR (SEA LEVEL)

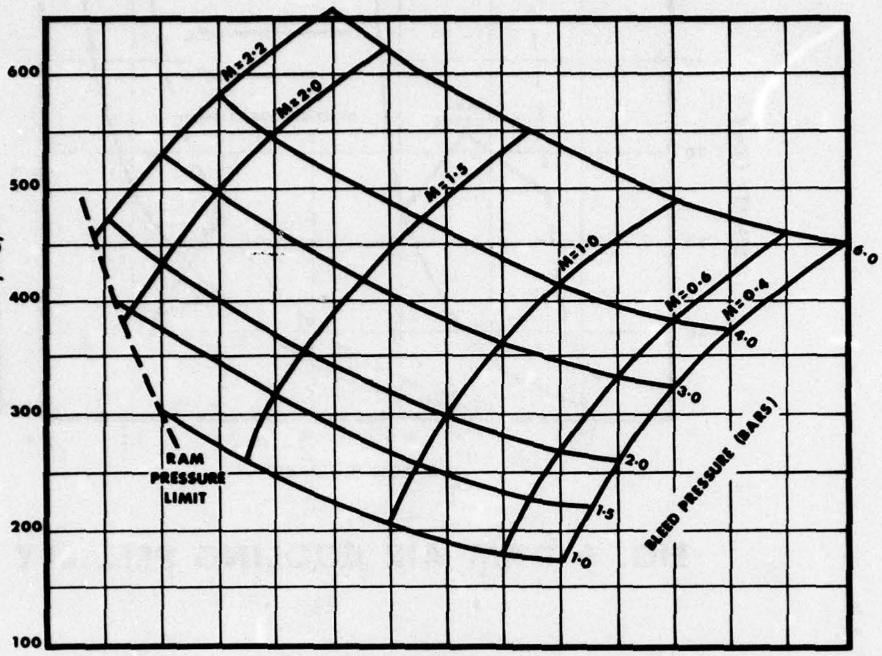


FIG. 4 PENALTY OF USING BLEED AIR (40000 ft.)

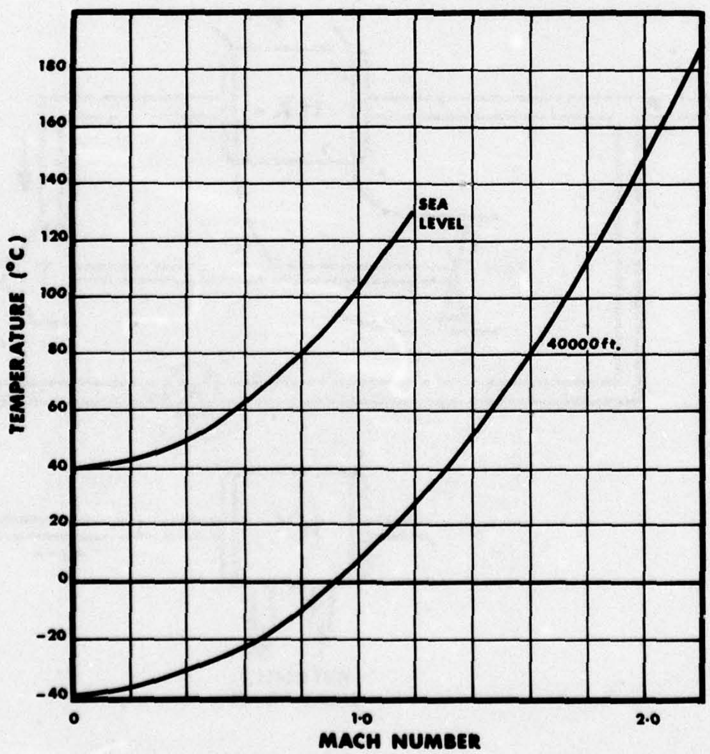


FIG. 5 TEMPERATURE OF RAM AIR

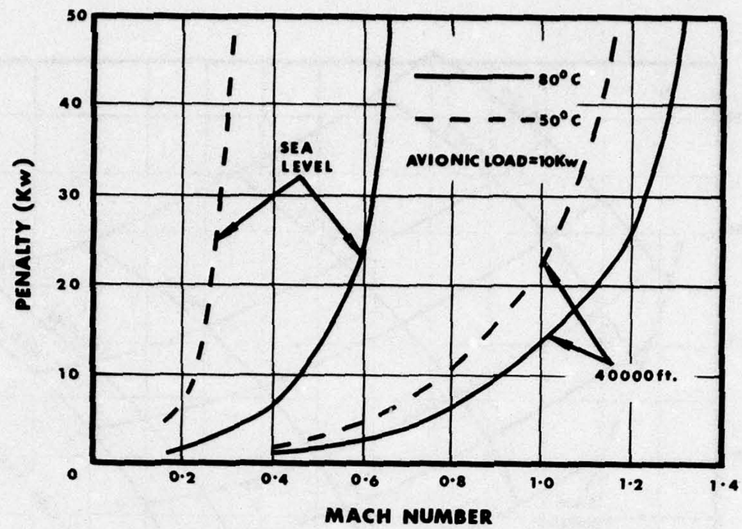


FIG. 6 RAM AIR COOLING PENALTY

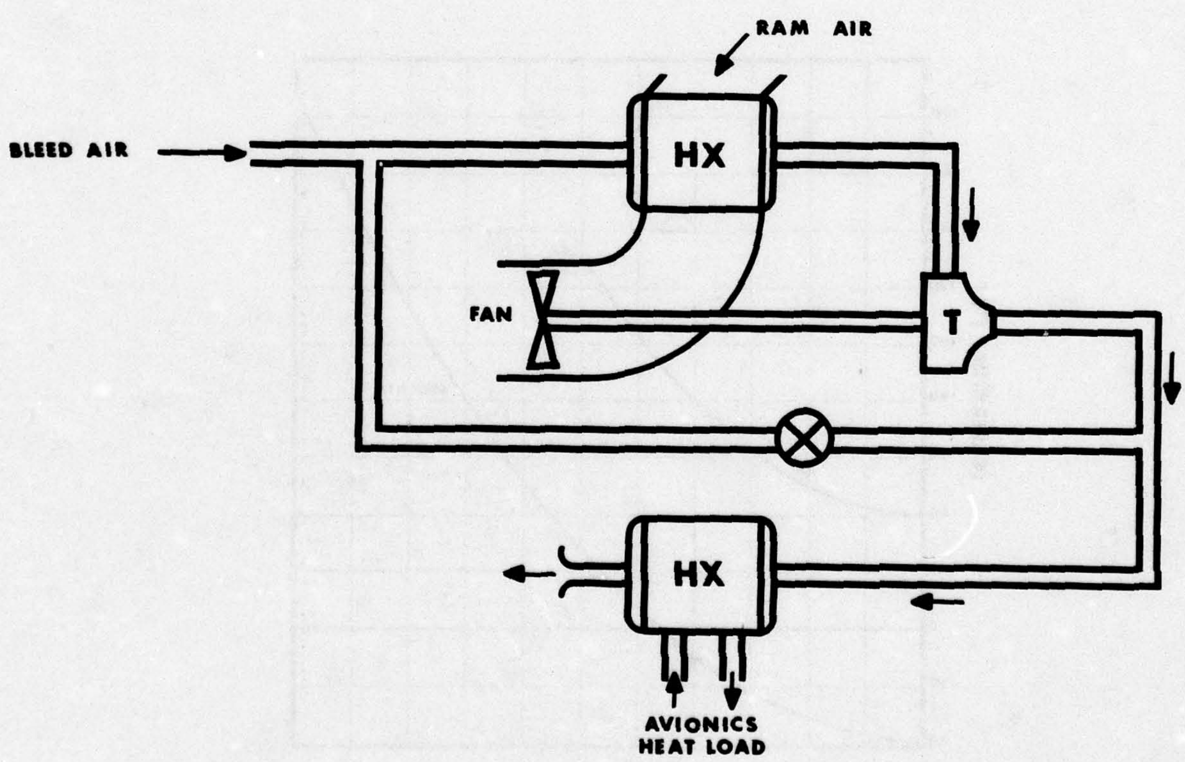


FIG. 7 TURBO–FAN COOLING SYSTEM

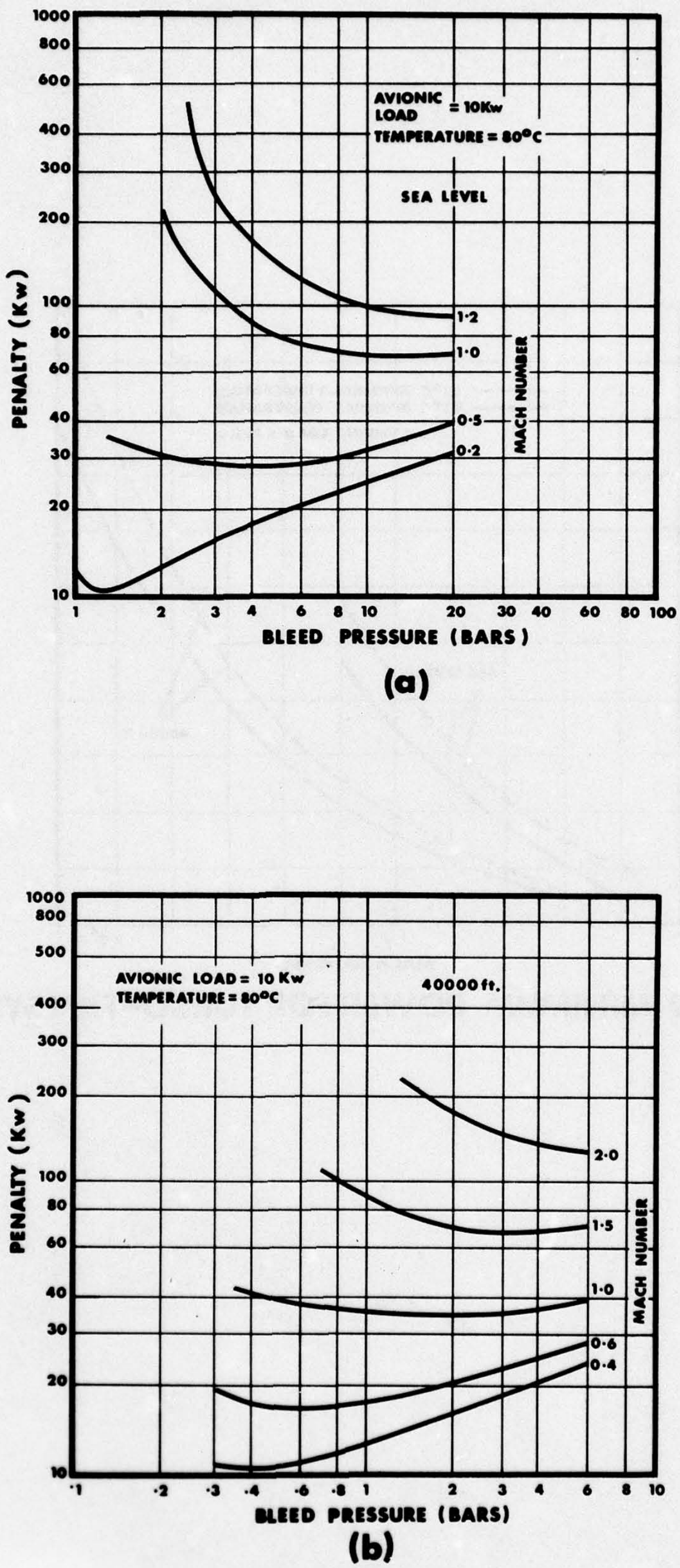


FIG. 8 POWER FOR TURBO-FAN SYSTEM

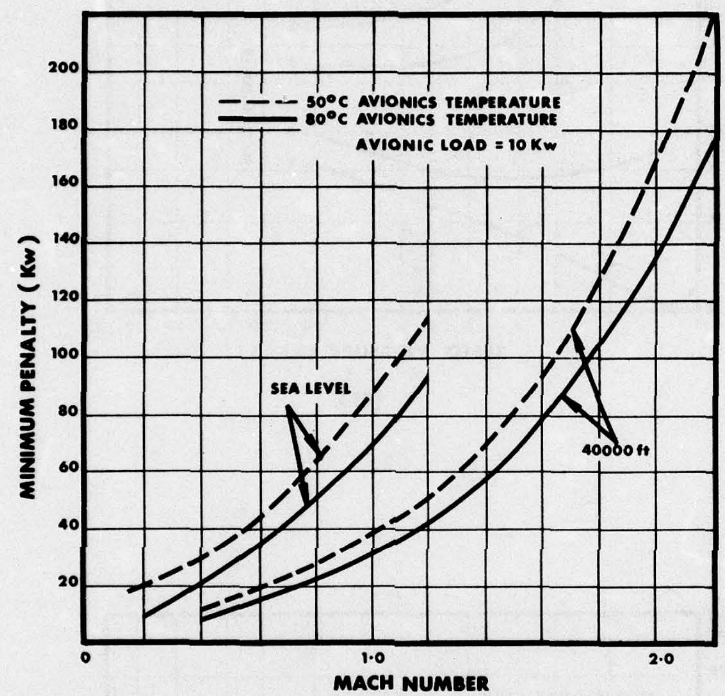


FIG.9 MINIMUM POWER FOR TURBO-FAN SYSTEM

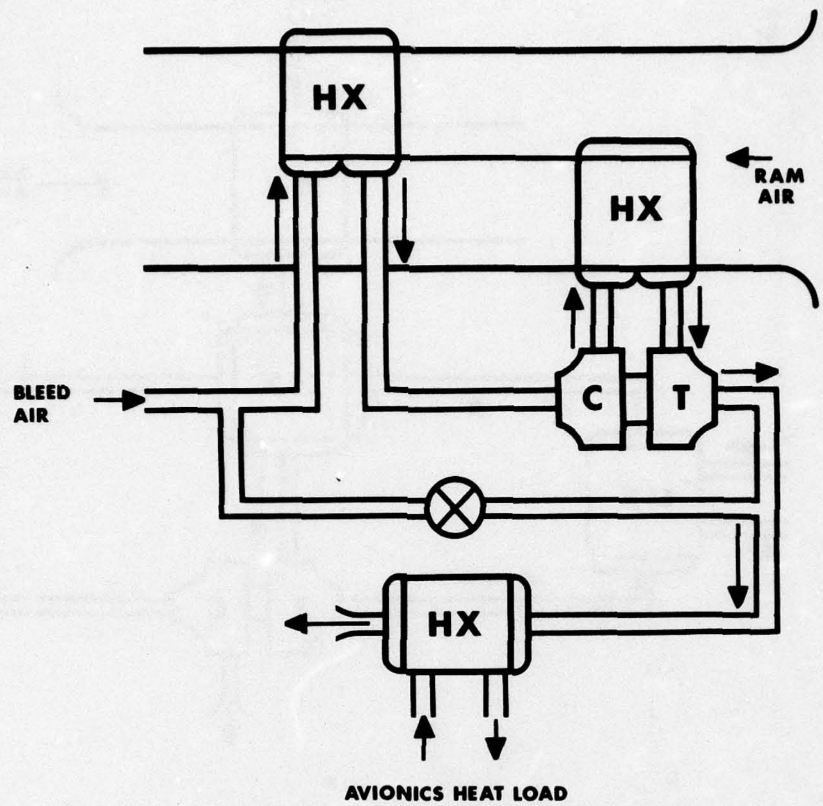


FIG. 10 BOOTSTRAP COOLING SYSTEM

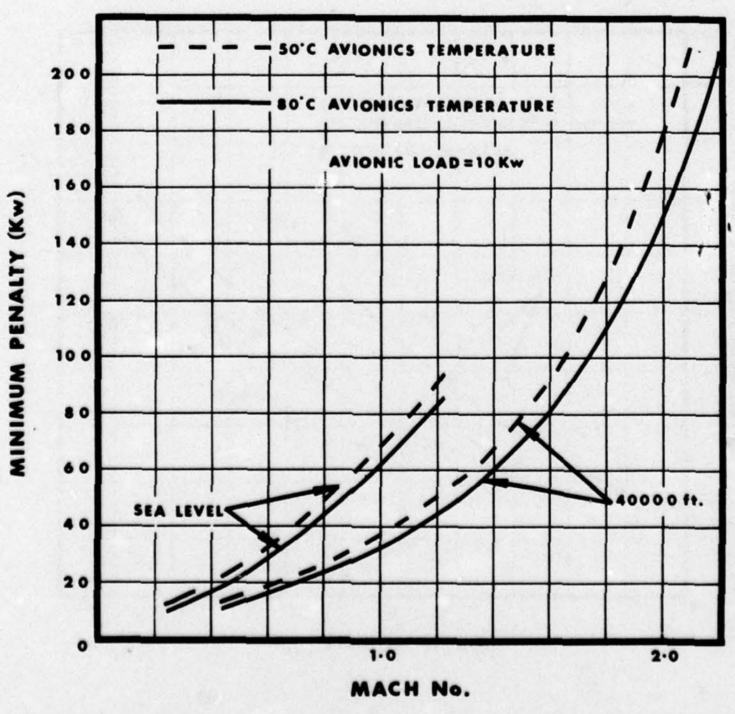


FIG. 11 MINIMUM POWER FOR BOOTSTRAP SYSTEM

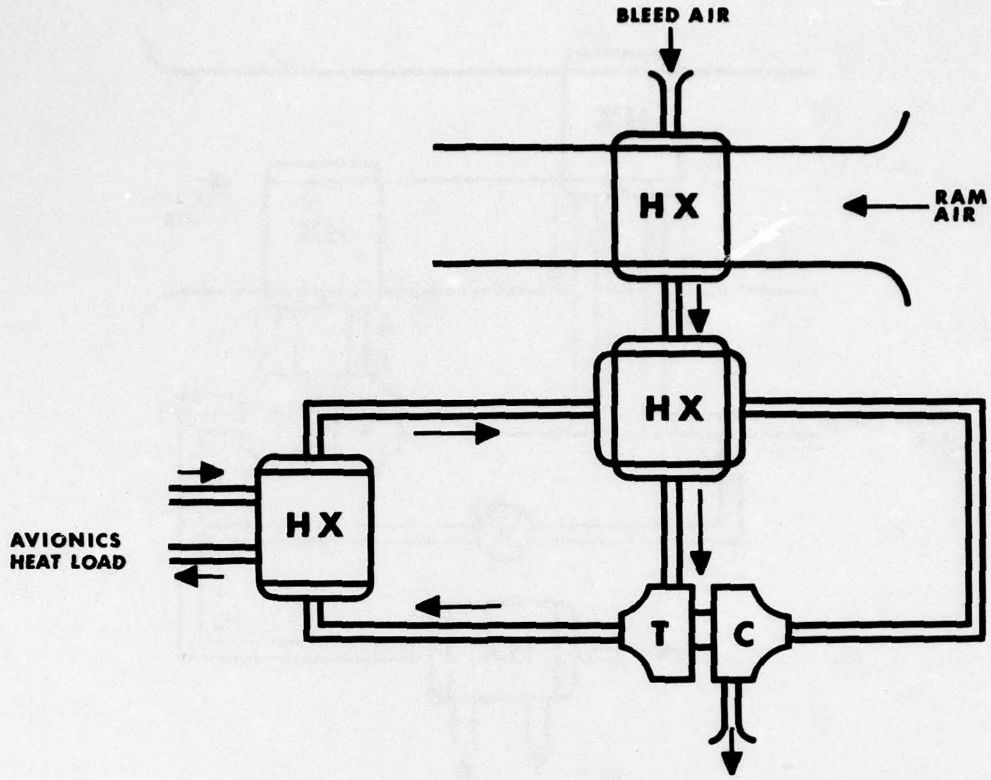


FIG. 12 REVERSED BOOTSTRAP SYSTEM

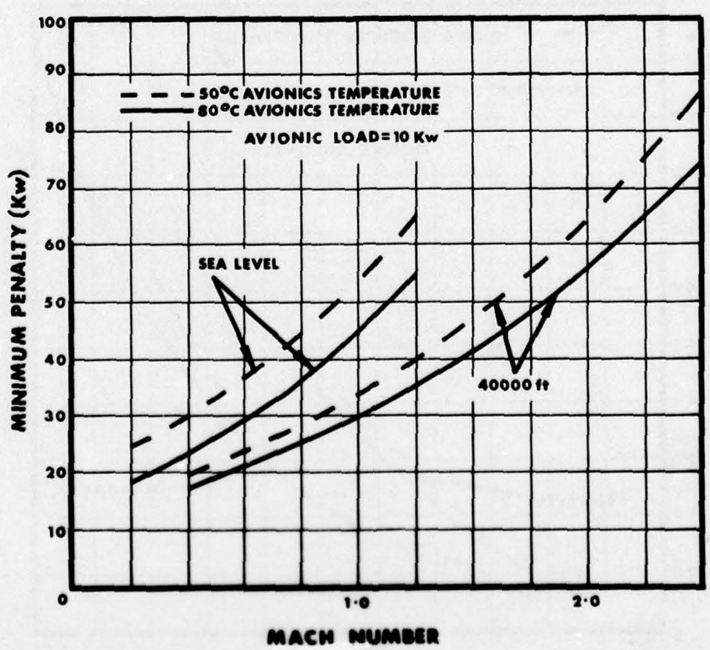


FIG. 13 MINIMUM POWER FOR REVERSED BOOTSTRAP SYSTEM

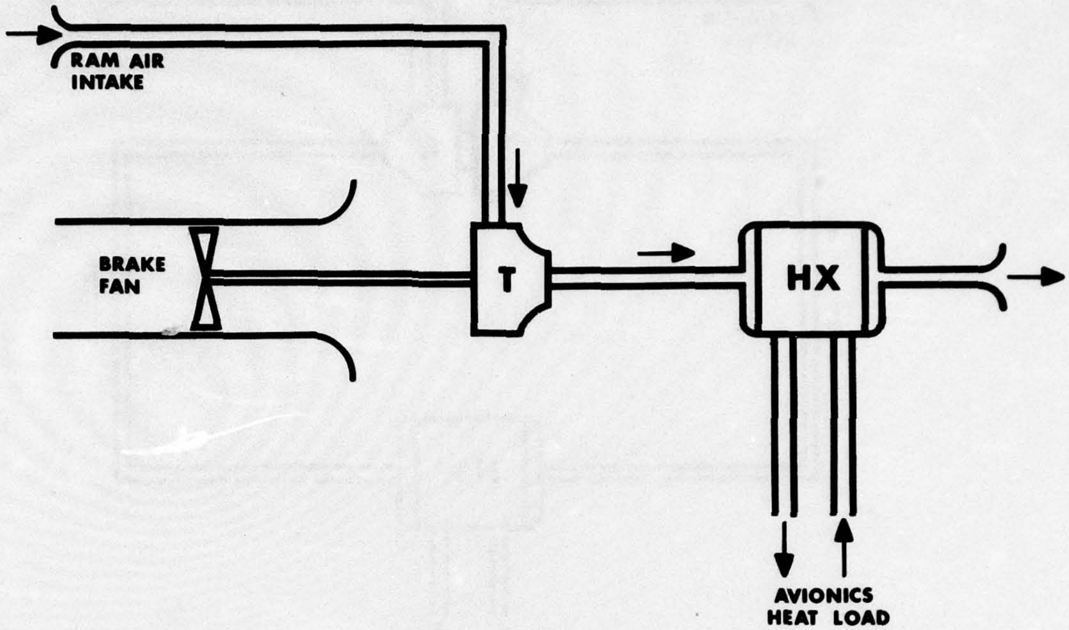


FIG. 14 EXPANDED RAM AIR SYSTEM

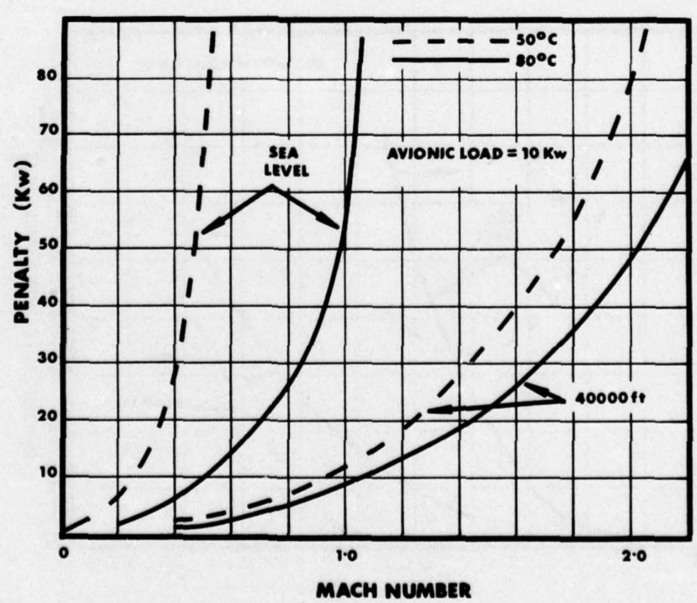


FIG.15 POWER FOR EXPANDED RAM AIR SYSTEM

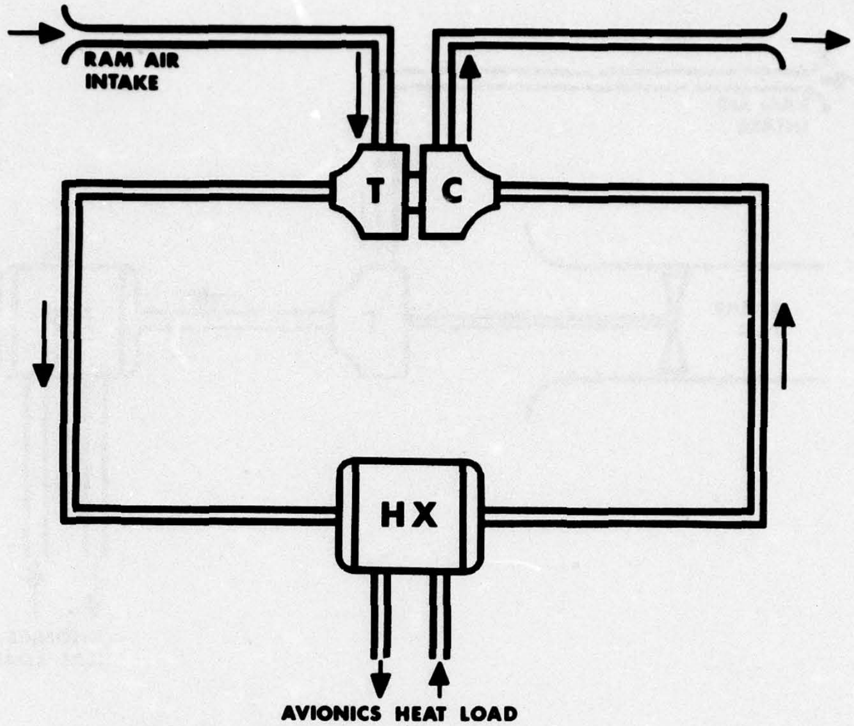


FIG.16 RAM POWERED REVERSED BOOTSTRAP

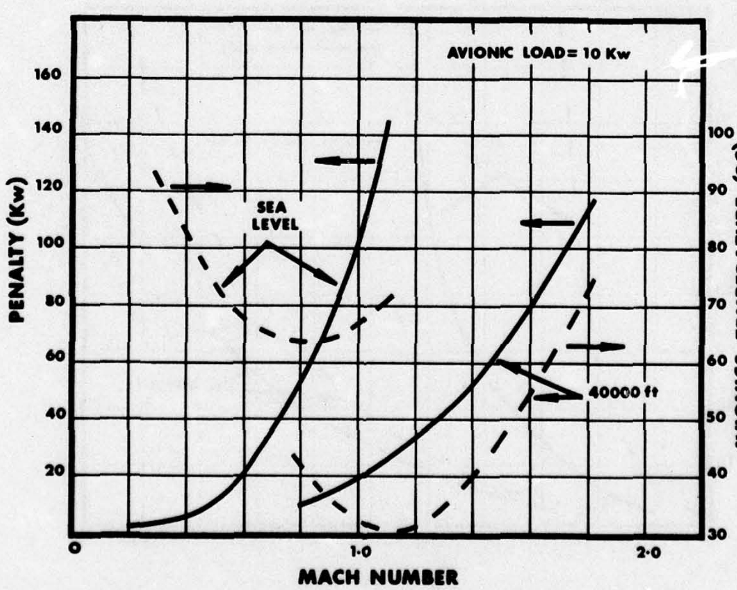


FIG.17 PERFORMANCE OF PRACTICAL RAM POWERED BOOT STRAP SYSTEM

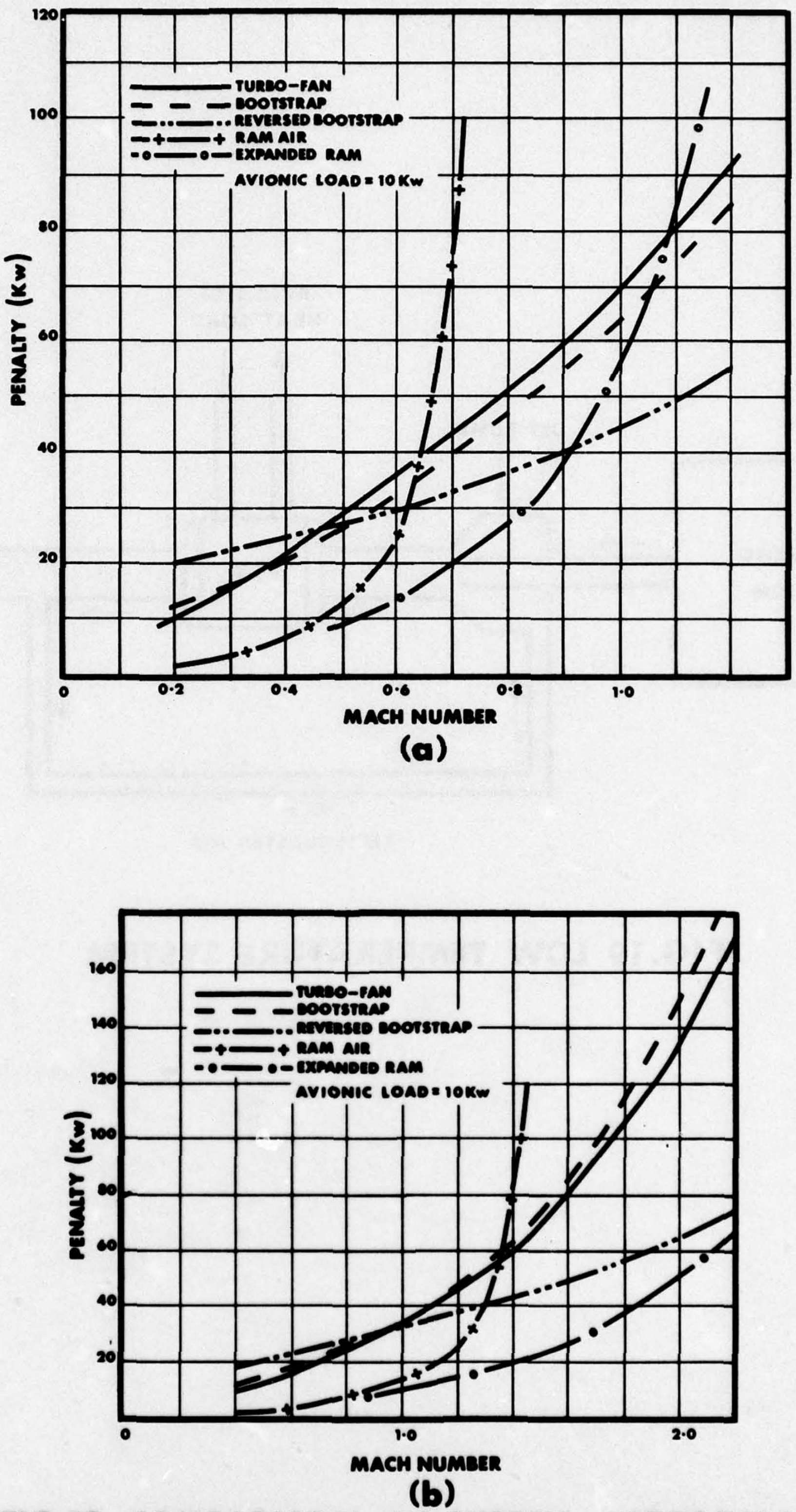


FIG.18. COMPARISON OF SYSTEM PERFORMANCE

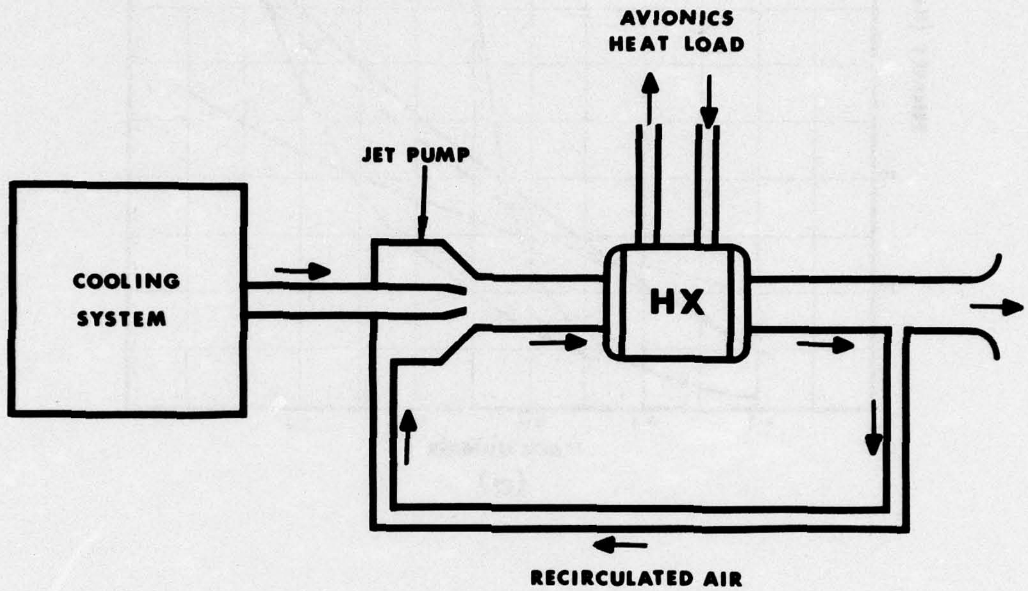


FIG.19 LOW TEMPERATURE SYSTEM

DISCUSSION

P Chapman:

Would a 25% reduction in avionic power reduce the penalties by 25%.

G F Stevenson and G R Giles:

Within reasonable limits this should be true. When everything is taken into account, eg weight, bulk of equipment and so on, then clearly some of these items only reduce as a square root function so that for very small systems the linearity will not be maintained.

P W Smith:

Your slide showed passengers requiring a temperature of 25°C stable to ± 1°. Whereas the requirement for avionics cooling is 50-70°C and no restriction on rates of change.

G F Stevenson and G R Giles:

Our paper deliberately set out to discuss the differences between Cabin and Avionic Cooling systems. The problems of obtaining reliability by lowering temperatures and holding them steady are understood and maybe we should have a set of requirements aimed at producing reliability in a similar way to the definition of physiological requirements of crew. A full overall study should consider the complete trade study between power penalty, reliability, maintainability and so on.

LE REFROIDISSEMENT DE L'EQUIPEMENT AVIONIQUE
A BORD DES AVIONS COMMERCIAUX

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RESUME

Cette étude essaye, par des exemples, de mettre en lumière les facteurs intervenant dans le refroidissement de l'équipement avionique à bord des avions commerciaux et présente les voies à suivre pour améliorer ce refroidissement de manière économique.

INTRODUCTION

LA CHALEUR, inévitable sous-produit de tout équipement électronique, apparaît de plus en plus comme l'ennemi implacable aussi bien pour l'architecture du circuit, pour l'avionneur qui l'installe, que pour l'utilisateur compagnie aérienne commerciale.

HISTORIQUE

Dès le début du transport aérien, vers 1936-1940, les compagnies aériennes et les constructeurs d'avions se sont penchés sur le problème de la rationalisation des dimensions des boîtiers de l'équipement radio-électrique, puis électronique utilisé à bord. C'est à cette époque qu'apparaît, au moins aux Etats-Unis, le style "ATR" de la spécification ARINC AS-11, dérivé des études d'UNITED AIRLINES pour le DOUGLAS DC-4.
A cette époque le refroidissement ne posait pas encore de problèmes : les équipements étaient largement espacés et peu nombreux.

La spécification ARINC AS-11, devenue ARINC 303, va servir de base à la définition des installations sur les avions nouveaux DOUGLAS DC-6, BOEING 377 Stratocruiser, LOCKHEED 049 Constellation, tandis qu'en Grande-Bretagne apparaît le système universel du SBAC (Society of British Aircraft Constructors), véritable "MECCANO" mais trop spécifique pour être adopté par d'autres que les Britanniques. Ce système disparaît avec le DE HAVILLAND "COMET".

C'est à cette époque que le problème du refroidissement commence à se faire jour : les équipements serrés comme sardines en boîte, souffrent et cuisent lentement. Leur fiabilité s'en ressent et le nombre des déposes s'accroît. Des essais entrepris par DOUGLAS en 1953 sur des ensembles montés dans des racks, montrent que les températures ambiantes sont beaucoup plus élevées que celles prévues et que celles obtenues lors des essais

d'ambiance effectués par les constructeurs. Equipementiers et avionneurs se renvoient mutuellement la responsabilité.

Les matériels continuent à évoluer, à se miniaturiser. Les transistors apparaissent et remplacent les tubes électroniques, gros générateurs de chaleur, le volume se réduit avec la taille des composants et loin de clarifier le problème, cette diminution du volume amène une dégradation de la situation.

La Spécification ARINC 404 de 1956 commence à donner des directives pour le refroidissement des équipements. Beaucoup de libertés sont laissées aux réalisateurs : refroidissement par convection libre, par air aspiré à travers l'équipement, par ventilateurs. Et les équipements continuent de souffrir, les utilisateurs de se plaindre, les Bureaux d'Etude d'essayer de résoudre les problèmes.

Une refonte de la Spécification ARINC 404 est à peine sous presse que déjà des récriminations se font jour. Même sur les avions gros porteurs, le problème du refroidissement est toujours aussi aigu. Cet état de fait entraine une réaction des Compagnies Aériennes pour l'étude des Concepts nouveaux d'installation (N.I.C.) dont le but est de définir, par un procédé évolutif plutôt que révolutionnaire, une spécification portant sur les interfaces : mécaniques, électriques et surtout d'ambiance entre les équipements électroniques, les étagères et armoires les recevant et les avions les utilisant, ceci économiquement et de façon à obtenir une espèrance de vie entre pannes se chiffrant en milliers d'heures.

EXPERIENCE CONCORDE

Cet avion (Photo 1) est le premier avion supersonique à entrer en service sur les lignes mondiales. Les problèmes rencontrés sur cet appareil sont plus importants que sur un avion subsonique.
En effet la vitesse de croisière élevée, Mach 2, produit un échauffement aérodynamique important des parois, environ 100°C, d'où la nécessité d'une isolation importante de celles-ci et d'un refroidissement énergique en croisière, alors que pour un avion subsonique les frigories ne font pas défaut à ce moment là. Les emplacements destinés aux armoires électroniques sont limités et le matériel y est très entassé.(Photos 2 - 3 - 4).

Les recommandations de la Spécification ARINC 404 : aspiration de l'air à travers l'équipement, ont été appliquées dans le dessin des étagères mais peu d'équipements sont prévus pour ce type de ventilation, la plupart acceptant un refroidissement par convection, du moins en théorie.

Chaque étagère dessinée spécialement, comporte donc une "boîte à vent", reliée au système d'évacuation de l'air. Sur celle-ci sont fixés des plateaux comportant une ouverture munie d'un diaphragme de réglage du débit d'air, et d'un joint d'étanchéité. (Photos 5 - 6).

La circulation de l'air se fait en évacuant l'air chaud vers l'extérieur, par la dépression régnant en altitude,ou par ventilateur au sol.

L'air de refroidissement est de l'air ambiant prélevé par un collecteur dans la cabine passagers : un ventilateur supplée au sol à la pressurisation cabine et en vol, en cas de panne, de façon à maintenir dans tous les cas un débit suffisant. (Planche 1-).

Il est à une température de 25 à 30°C et un filtre est monté dans chacune des canalisations (droite et gauche) de prélèvement en cabine. Le débit d'air nécessaire pour refroidir chaque meuble est d'environ 680 $kg.h^{-1}$ (soit $\simeq$ 0,2 $kg.s^{-1}$) pour une puissance électrique dissipée d'environ 4 kW. Cette valeur est de l'ordre de celle normalement allouée

par la spécification ARINC 404 (136 $kg.h^{-1}.kW^{-1}$) et très inférieure à celle des spécifications ARINC 404 A et ARINC 600 (220 $kg.h^{-1}.kW^{-1}$).
Cependant l'ensemble des équipements, des galleys, et des panneaux du poste de pilotage nécessite un débit d'air d'environ 3720 $kg.h^{-1}$. Le débit normal d'air de conditionnement cabine est d'environ 4900 $kg.h^{-1}$. Il ne reste donc que peu d'air de fuite de cabine vers les soutes pour y maintenir une ambiance acceptable.

EXPERIENCE AIRBUS

L'AIRBUS, A300B, (Photo 7) est un avion gros porteur, moyen courrier subsonique dont la définition est plus récente que celle de CONCORDE.
Etant donné l'espace disponible dans la partie inférieure avant de l'appareil, un principe différent de la norme ARINC 404 a été utilisé. (Photos 8 - 9 - 10).

Les équipements sont simplement refroidis par convection. L'air ambiant et chaud est collecté au-dessus de chacune des étagères supportant les boîtes noires et est rejeté vers l'extérieur soit par dépression en vol soit par un ventilateur au sol. (Planche 2). L'air ambiant est renouvelé par les "fuites" de cabine. Cet air est donc aussi relativement chaud et pollué.
Le débit d'air dans le meuble droit est de 280 $kg.h^{-1}$ (80 $g.s^{-1}$) pour 1600 W et dans le meuble gauche de 650 $kg.h^{-1}$ (180 $g.s^{-1}$) pour 2500 W. Ces valeurs sont peu supérieures à celles préconisées par l'ARINC 404 A et permettent en vol normal d'obtenir une température de fonctionnement correct.
Au sol, par temps chaud un soufflage d'air frais, provenant du circuit de conditionnement du poste de pilotage est effectué à la partie inférieure de chaque étagère.

Ces dispositions ont l'avantage de procurer un refroidissement correct en général en évitant les deux inconvénients de la ventilation type ARINC par aspiration :

1° - Les étagères peuvent n'être que de simples plateaux supports, sans joint d'étanchéité toujours sujet à détérioration et dont la disposition peut être banalisée.

2° - Les pollutions (nicotine, poussières) ne se déposent pas à l'intérieur des équipements électroniques.

En résumé, le refroidissement des équipements électroniques : planches de bord, meuble disjoncteurs, nécessite un débit d'air de 2800 $kg.h^{-1}$ (600 $g.s^{-1}$) à comparer avec les 7900 $kg.h^{-1}$ (2200 $g.s^{-1}$) nécessaires au conditionnement d'air de la cabine.

PROJET NIC

En 1972, alors que la spécification ARINC 404 en était au stade final de sa rédaction, une proposition fut faite envisageant des Concepts Nouveaux d'Installation (N.I.C.) ayant les objectifs suivants :

- Fiabilité augmentée
- Maintenance facilitée
- Cablâge simplifié
- Flexibilité de modification
- Compatibilité avec la spécification ARINC 404 A
- Coûts diminués

Une maquette très évoluée était présentée par DOUGLAS. (photo 11).

Du fait de son avance technologique, elle a soulevé de gros remous particulièrement parmi les utilisateurs, effrayés par cette révolution. Une évolution plus lente a été décidée et mise au point.

La première phase de cette évolution est pratiquement acquise : et ainsi que le disait le Dr J.V.N.GRANGER du Stanford Research Institute en exergue de la spécification ARINC 404 de 1956 = "Old Operational Requirements never die. They just shrink to 1/4 ATR size". Il est maintenant proposé un module de base 1MCU identique au 1/8 ATR !!! "So they just shrink to 1/8 ATR now !!!..."

Un comité et des sous-comités ont été créés aussi bien en EUROPE qu'aux ETATS-UNIS, pour étudier les problèmes, et des maquettes ont été réalisées. (Photo 12).
Deux points particuliers ont été retenus pour cette première phase :

- Connecteur à faible force d'insertion. Le connecteur actuel, du fait du nombre toujours plus grand de contacts, entraine des efforts importants, à l'insertion et à l'extraction, source de déformations, d'où faux contacts, broches tordues, difficultés de mise en place, etc ...
- Amélioration de l'environnement surtout du point de vue thermique et c'est là le point principal de ces propositions.

Pour fonctionner dans des conditions satisfaisantes et avec une fiabilité atteignant des milliers d'heures, les équipements électroniques doivent être refroidis pour évacuer toutes les calories générées par le fonctionnement.

La tendance actuelle à la miniaturisation des circuits électroniques : intégration de nombreuses fonctions sur une seule pastille de semi-conducteur, entraine des dissipations par unité de volume importantes. De plus il est difficile d'évacuer la chaleur produite pratiquement en un point.

Dans cette première phase, l'air seul est retenu comme moyen principal pour refroidir les boîtes noires. Un débit important : 220 $kg.hr^{-1}.kW^{-1}$; est prévu en même temps que la dissipation est limitée à 25 W pour le module unitaire 1 MCU.
Il est significatif de comparer cette limitation de dissipation et de débit avec des valeurs adoptées récemment (MIAMI Déc.1975) par exemple pour la spécification ARINC 559 A pour l'émetteur de communication HF-SSB.
Pour un volume 3/4 ATR court, qui est équivalent à 6 modules unitaires, il est prévu une dissipation moyenne de 250 W (60 W en position Réception, 600 W en position Emission) soit 1,7 fois plus forte. Un débit d'air standard de 54 $kg.h^{-1}$ est prévu mais certains fabricants ont déjà demandé le double soit : 108 $kg.h^{-1}$ afin d'améliorer la fiabilité.

Cette limitation de la dissipation unitaire conduira à repenser les méthodes de refroidissement. Pour le moment l'air utilisé est en général l'air provenant de la cabine passagers donc pollué : poussières et fumées de tabac. Ces souillures ont une tendance fâcheuse à se déposer sur les parties les plus sensibles des circuits électroniques ! Il serait donc recommandable sinon indispensable de prévoir une épuration de cet air.
Pour les poussières, une élimination par action mécanique peut être envisagée : centrifugation par exemple, avec élimination automatique des déchets.
Par contre, pour les fumées de tabac il n'existe que peu de moyens simples : le filtre (laine de verre par exemple) semble en être un, avec cependant tous les inconvénients qu'il procure : colmatage plus ou moins rapide, d'où la nécessité d'une visite et d'un nettoyage périodique. La condensation sur paroi froide est aussi évoquée. De plus cette épuration produit des pertes de charges d'autant plus fortes qu'elle est d'autant plus poussée. Un moyen mécanique est nécessaire pour forcer l'air à travers l'épurateur ce qui l'échauffe et va à l'encontre du but recherché. En effet, les conditions d'ambiance auxquelles peuvent être soumis les matériels avioniques sont définis par un document commun

à la "Radio Technical Commission For Aeronautics" (R.T.C.A.) américaine et à l'Organisation Européenne pour l'Equipement Electronique de l'Aviation Civile (EUROCAE), européenne: RTCA DO.160/EUROCAE 1/WG14/75.

Ce document définit les limites haute et basse dans différents cas - en vol et au sol - la limite haute étant généralement la plus gênante : + 55°C normal ; + 70°C, température limite pendant 30 minutes. Ces valeurs élevées obligent les fabricants d'équipement à détarer les composants pour obtenir une fiabilité correcte en les faisant travailler à une température interne convenable.

Néanmoins, il est nécessaire d'étudier les équipements pour tirer le meilleur parti de la ventilation prévue. Elle n'est pas encore clairement définie par la spécification ARINC 600 mais les libertés de la spécification ARINC 404 A : refroidissement par convection et éventuellement ventilateurs, ont été supprimées.

Les études menées par l'AEROSPATIALE ont conduit à proposer pour la spécification ARINC 600 une disposition modulaire des ouvertures pour le passage de l'air de ventilation (planche 3).

Cette disposition, tout en laissant libre le choix du sens de passage de l'air permet de choisir :

- la ventilation de l'ensemble, ou d'une partie seule, de l'intérieur du boîtier l'échange de chaleur se faisant directement au niveau des composants calorifiques.
- pour un matériel de haute fiabilité, la ventilation d'un échangeur de chaleur, seul soumis, à ce moment là, au dépôt d'éléments polluants.
  La chaleur produite au niveau des composants est conduite à l'échangeur soit par le circuit imprimé à âme métallique soit par des caloducs (Heat Pipes en Anglais) (planche 4-)
- et même pour un "design" très raffiné, la ventilation des cartes creuses de circuit imprimé.

Ceci pour la ventilation proprement dite des boîtiers électroniques. Pour ce qui est du schéma de ventilation des étagères et meubles abritant ces boîtiers, plusieurs méthodes ont été étudiées : (planche 5)

<u>CAS 1</u> Aspiration directe à travers les boîtiers, facile à adapter mais inconvénients dus aux poussières et goudrons de tabac.

<u>CAS 2</u> Soufflage direct d'air mais dispersion des fumées (en cas de court-circuit interne) dans l'avion par recirculation.

<u>CAS 3</u>, <u>CAS 4</u>) Circulation forcée évitant les deux inconvénients précédents.

<u>CAS 5</u>, <u>CAS 6</u>) Circulation forcée dans une enceinte fermée.

Le cas 5 semble préférable à l'heure actuelle, car il permet :

- éventuellement de refroidir l'air avant soufflage dans l'enceinte
- d'accepter les équipements définis suivant la spécification ARINC 404 (sans ventilation), la spécification ARINC 404A(ventilation possible par aspiration) et la spécification ARINC 600.

Il faut signaler que l'air utilisé est à température élevée en <u>théorie</u> : + 55°C limite haute permanente, + 70°C, limite haute de courte durée, 30 minutes, et aussi en <u>pratique</u>. Ces valeurs limites sont souvent des valeurs moyennes.

Un remède à cet inconvénient serait la réfrigération de l'air de refroidissement, réfrigération qui serait très favorable pour la fiabilité.
Un bilan estimatif a été fait pour un avion tel que CONCORDE : une puissance de 40 kW et une masse de 500 kg pour fournir l'air nécessaire à une température de 10°C, seraient à prévoir. Le bilan est donc lourd pour l'avion : en poids, en énergie consommée et donc en chaleur à évacuer.

Cette solution n'a pas encore été envisagée pour les avions futurs et à l'heure actuelle les équipements doivent être prévus et certifiés à ces ambiances. La spécification ARINC 600 propose une méthode d'essais devant servir de base pour la qualification thermique des équipements selon des critères acceptés par tous : équipementiers, avionneurs, compagnies aériennes utilisatrices et services officiels, critères donnant une fiabilité élevée pour un coût minimal.

Pour s'assurer de cette certification,chaque équipement, au moins durant sa phase de développement et peut-être aussi durant la phase de mise en service, devrait comporter un élément sensible à la chaleur. Ce serait un genre de "B.I.T.E." (Built In Test Equipment) permettant à l'avionneur et à l'utilisateur, de contrôler si les conditions de fonctionnement prévues par l'équipementier sont respectées. Les discussions, où chacun se renvoie la responsabilité du mauvais fonctionnement, pourraient être évitées ainsi que les modifications toujours coûteuses sur les avions en service.

CONCLUSION

A bord des avions commerciaux, le refroidissement de l'équipement avionique a toujours présenté des problèmes.

Un compromis entre différents facteurs est nécessaire :

- Fiabilité
- Prix de revient
- Complexité
- Poids
- Facilité d'entretien

Ce compromis doit donc être le fruit de la coopération entre les trois maillons de la chaîne :

- Equipementier
- Avionneur
- Utilisateur

Les nouveaux concepts d'Installation NIC proposent une voie pour améliorer les problèmes résultant de la complexité croissante des Equipements Electroniques.
La première phase arrive à sa mise au point finale.

Les phases suivantes permettront, par une évolution progressive, d'aboutir à des

solutions satisfaisantes par l'emploi des techniques nouvelles, développées parfois pour l'utilisation spatiale : refroidissement par conduction ; par liquide et même par ébullition. Un gros effort reste à faire dans ce domaine pour adapter ces techniques nouvelles aux exigences de l'utilisation sur avions commerciaux : facilité d'emploi, manipulation aisée, entretien simple, prix de revient acceptable.

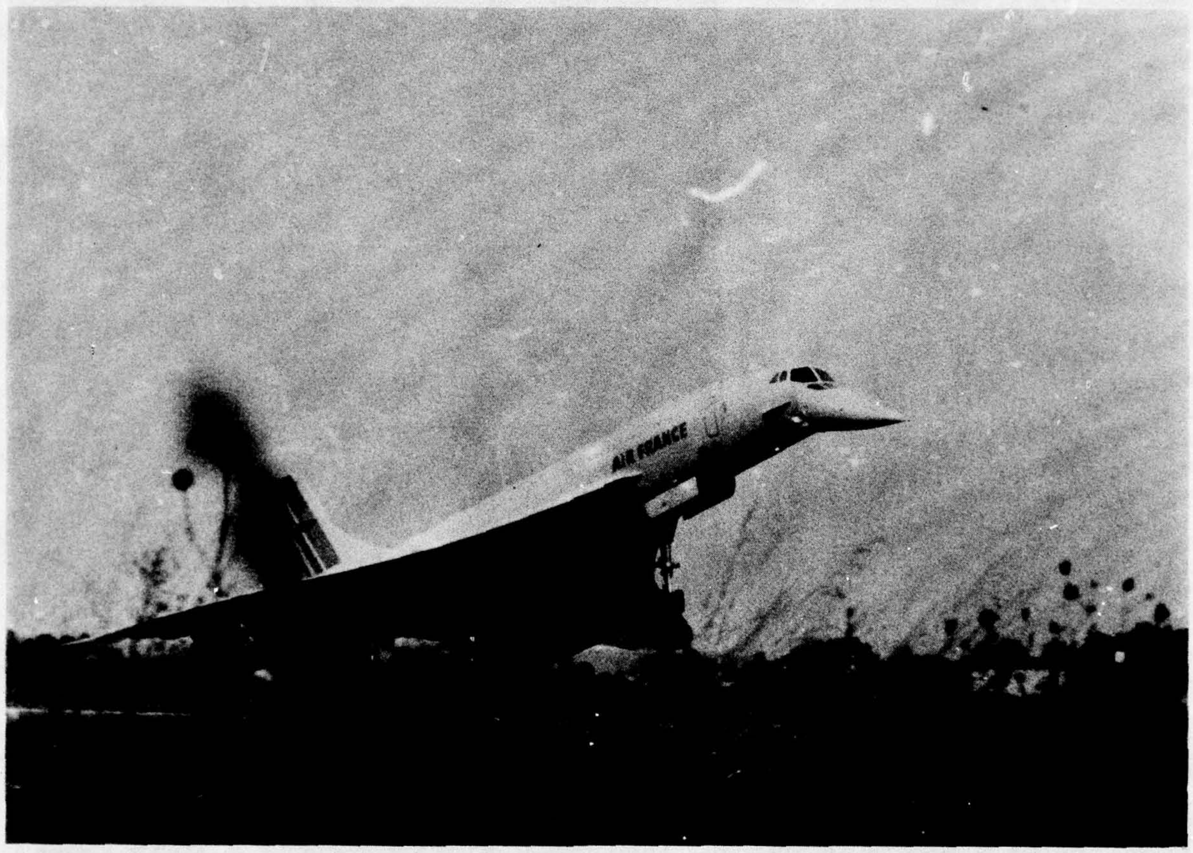

Photo 1 CONCORDE au décollage

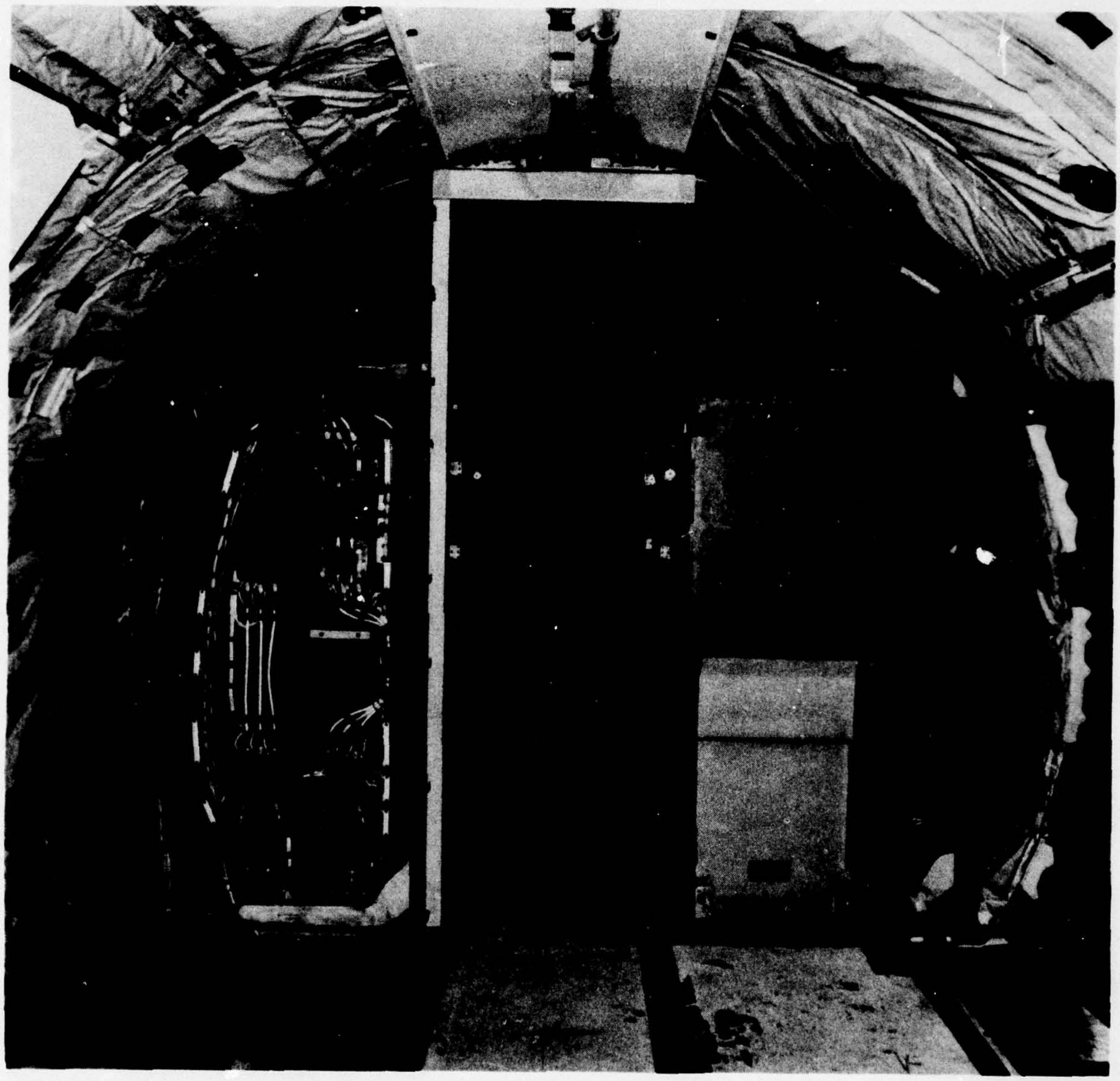

Photo 2 CONCORDE – Vue vers l'avant
Les armoires électroniques sont situées à droite et à gauche du couloir allant au poste de pilotage

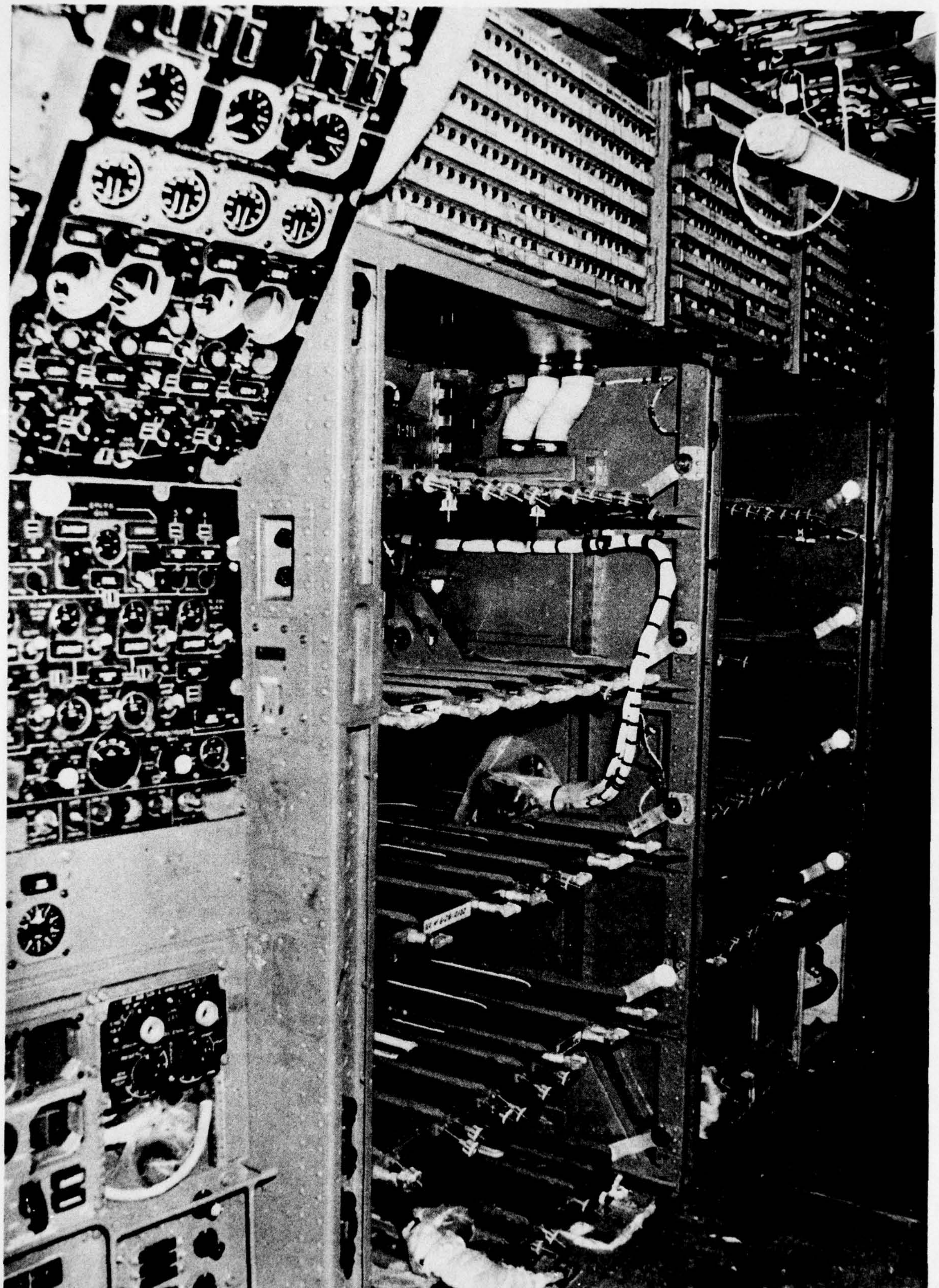

Photo 3 CONCORDE – Armoire droite
Vue du poste de pilotage

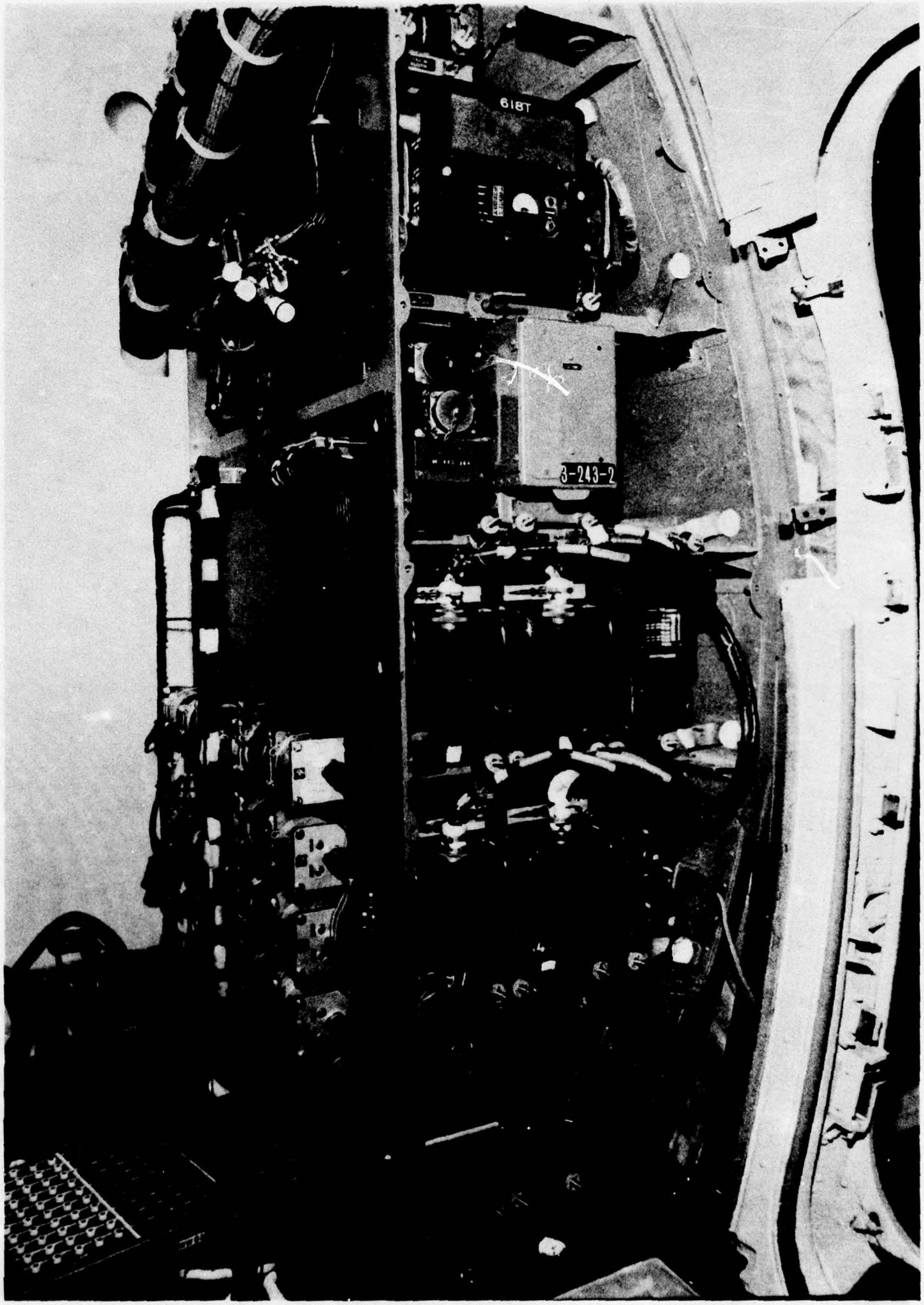

Photo 4 CONCORDE – Armoire gauche arrière

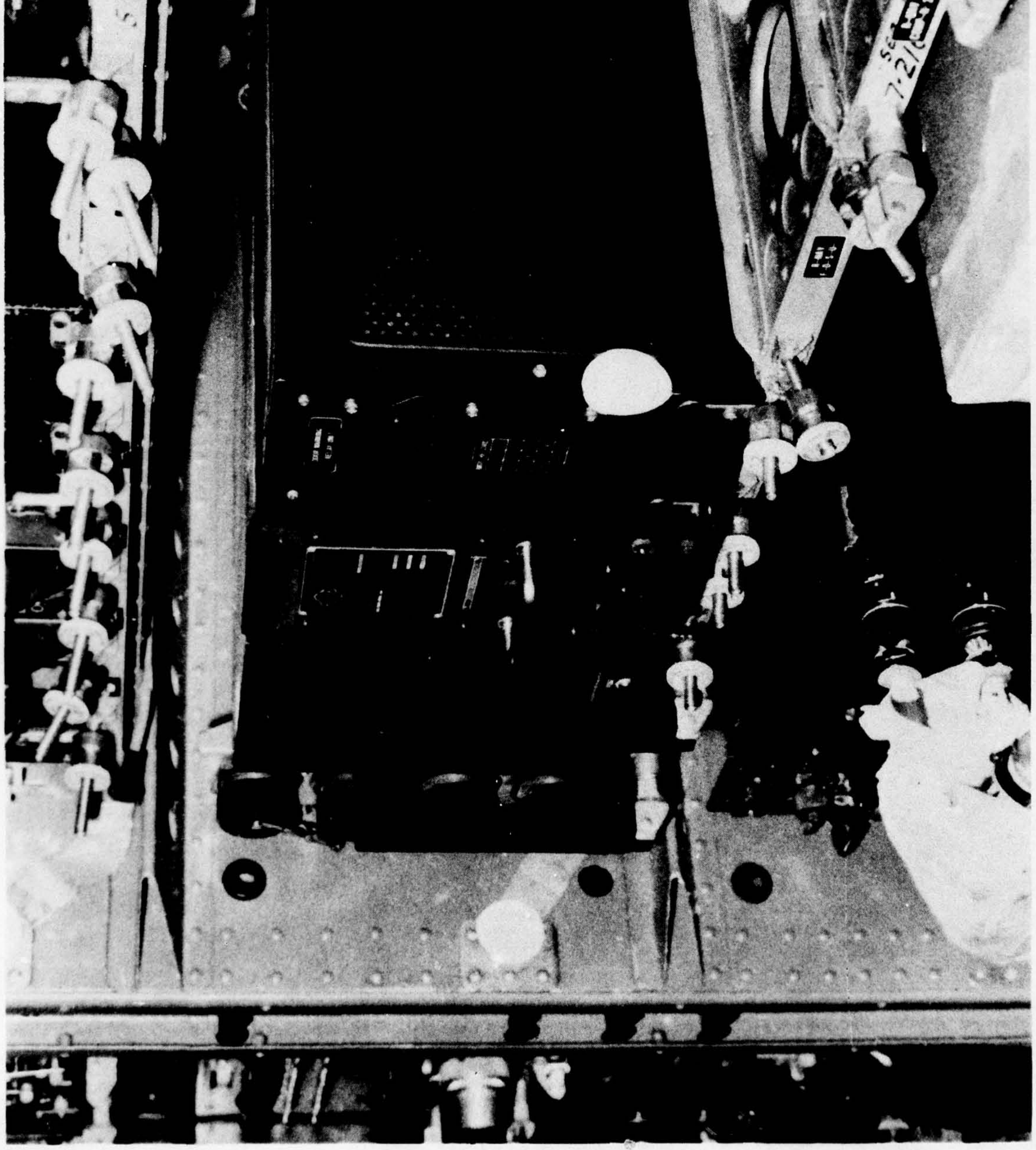

Photo 5   CONCORDE – Etagère ventilée

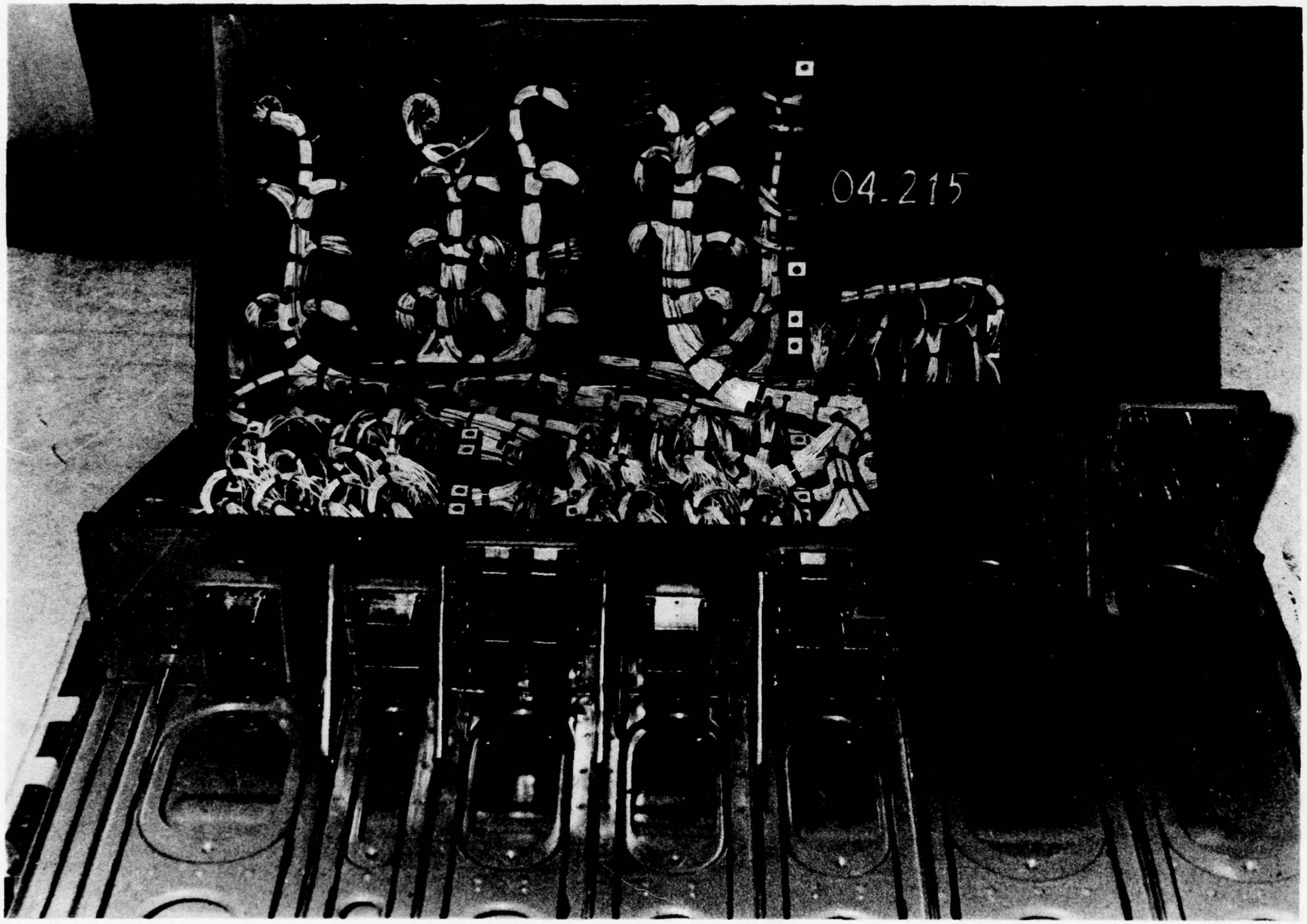

Photo 6 CONCORDE – Etagère ventilée

aérospatiale

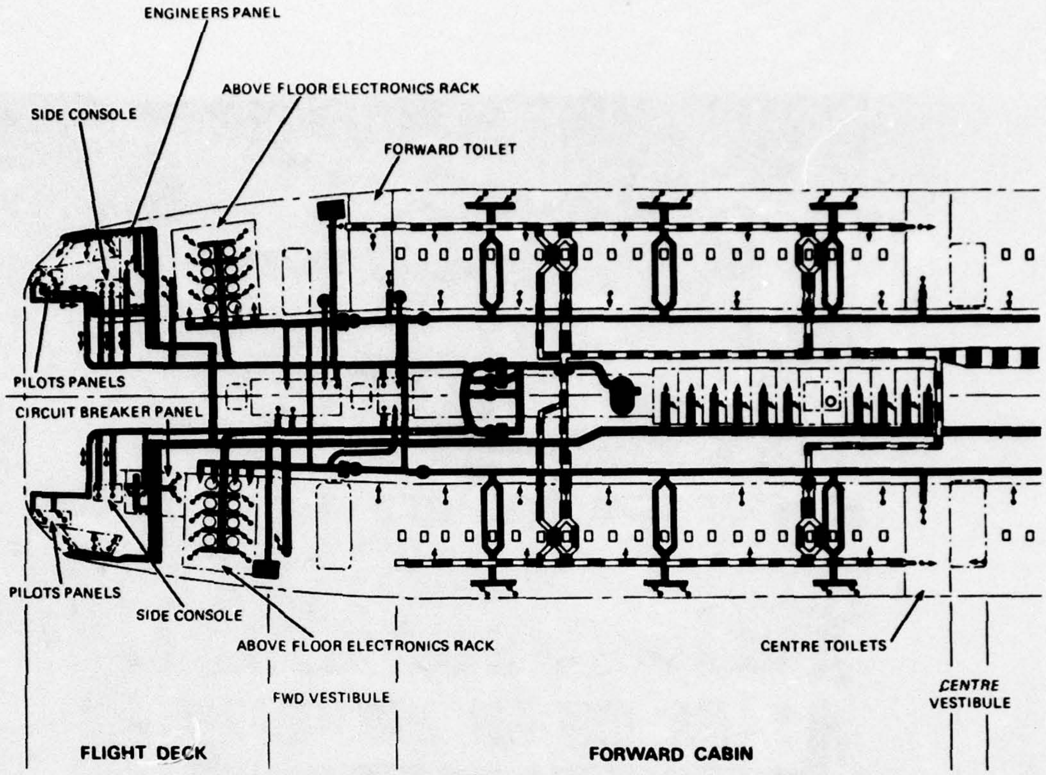

Planche 1 T.S.S. CONCORDE – Avionics ventilation and extraction

Photo 7 AIRBUS

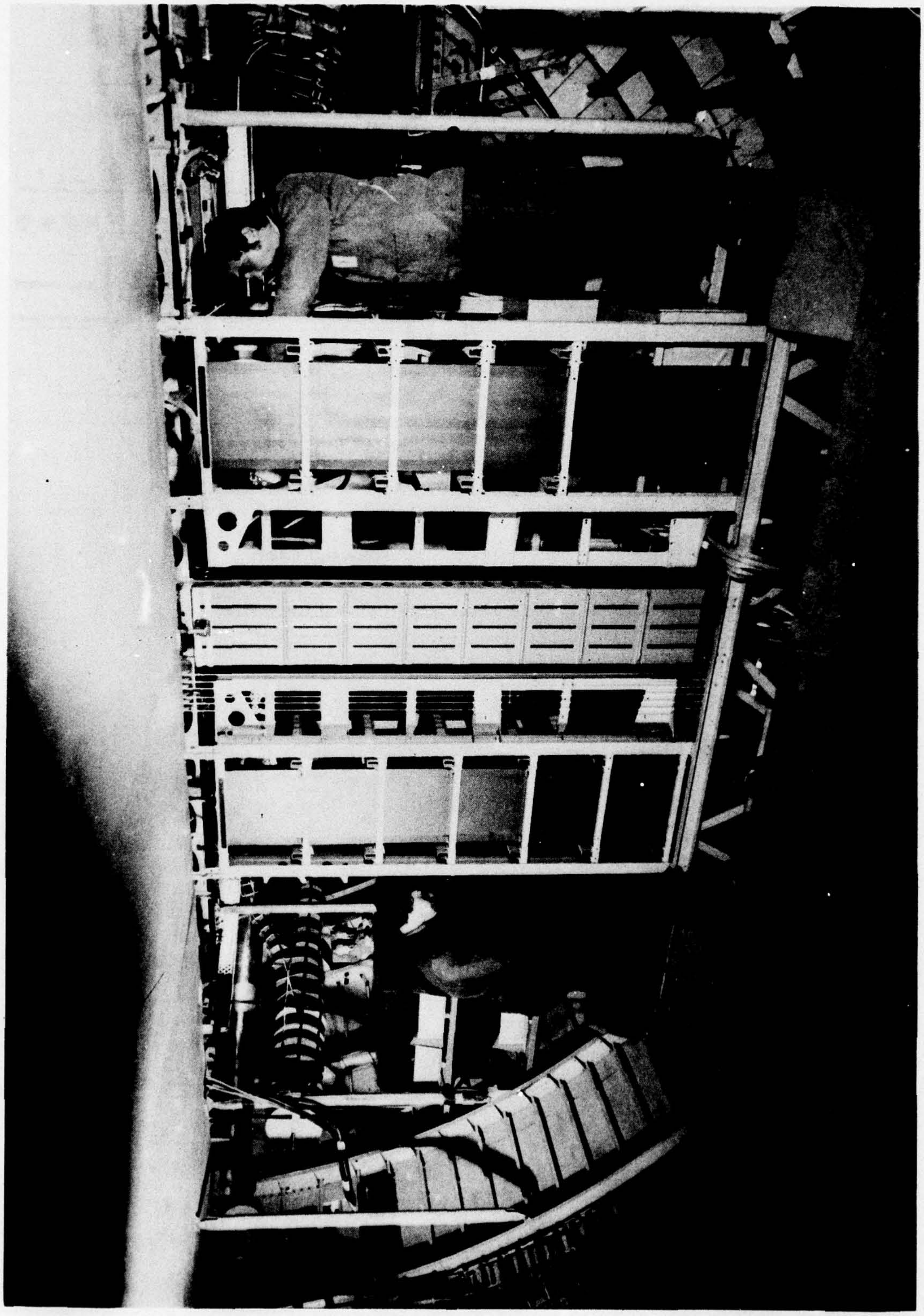

Photo 8 AIRBUS – Soute électronique
Les armoires électroniques sont situées au centre – Vue vers l'avant

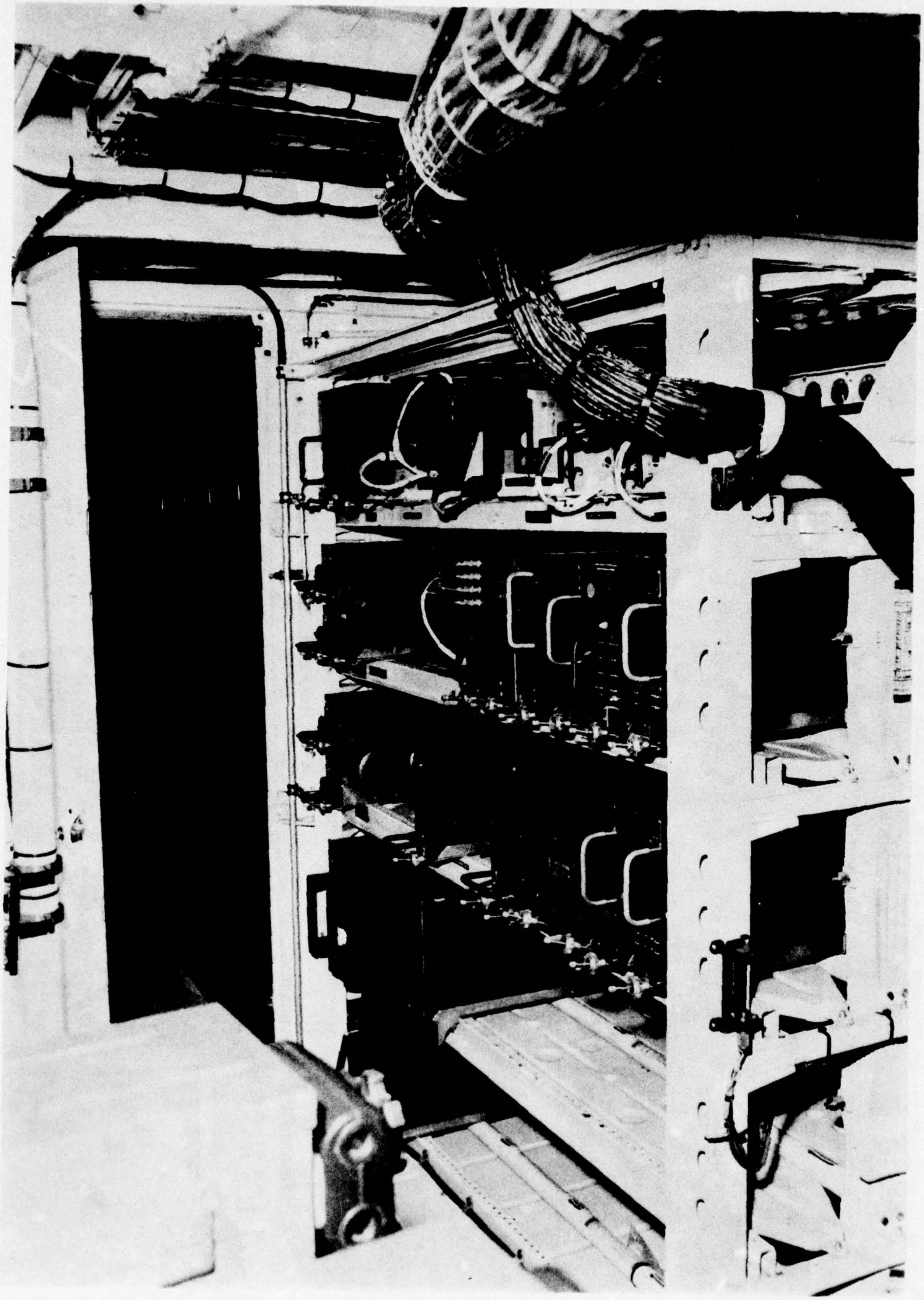

Photo 9 AIRBUS – Armoire droite
Vue vers l'arrière

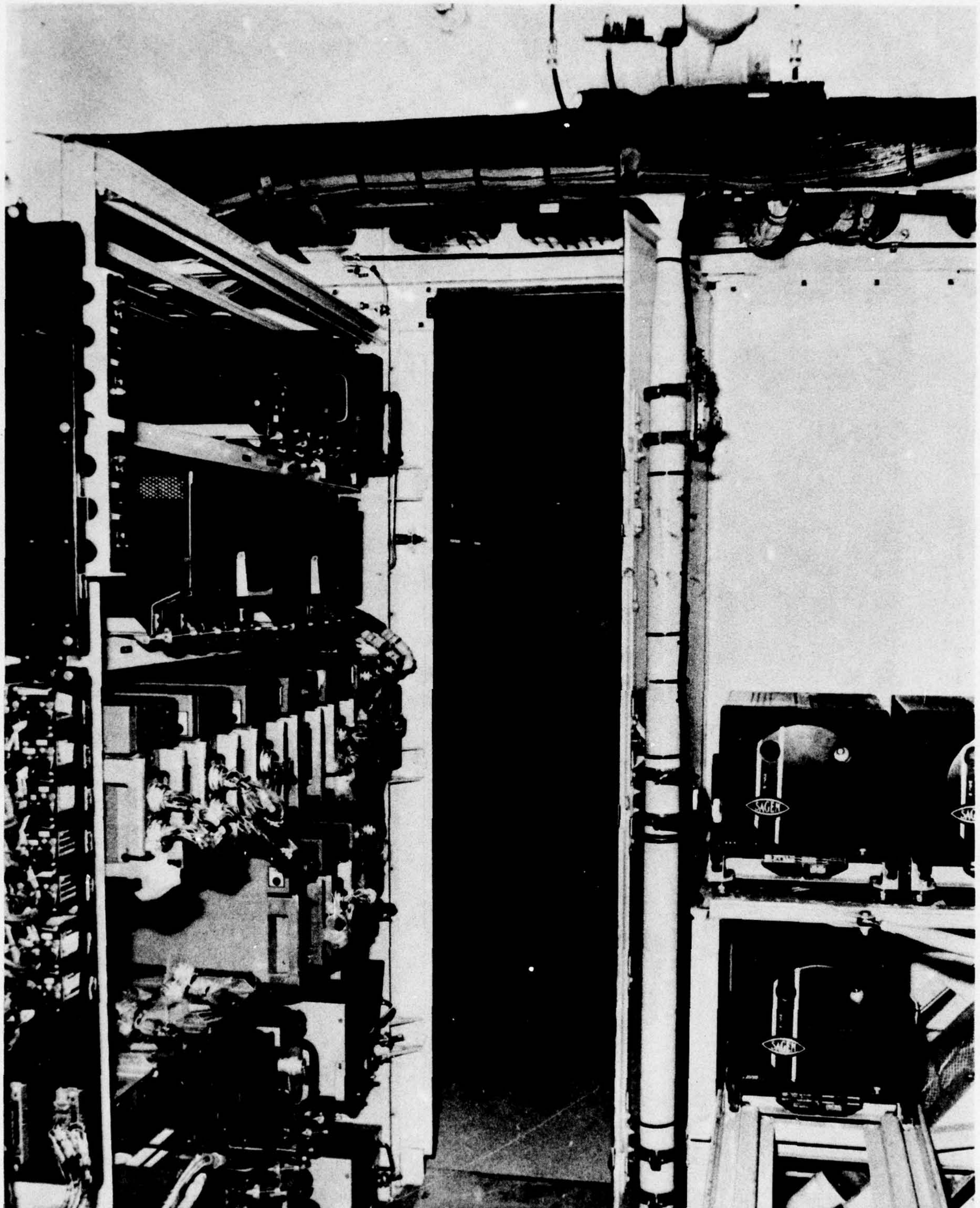

Photo 10 AIRBUS – Armoire gauche
Vue vers l'arrière

aérospatiale

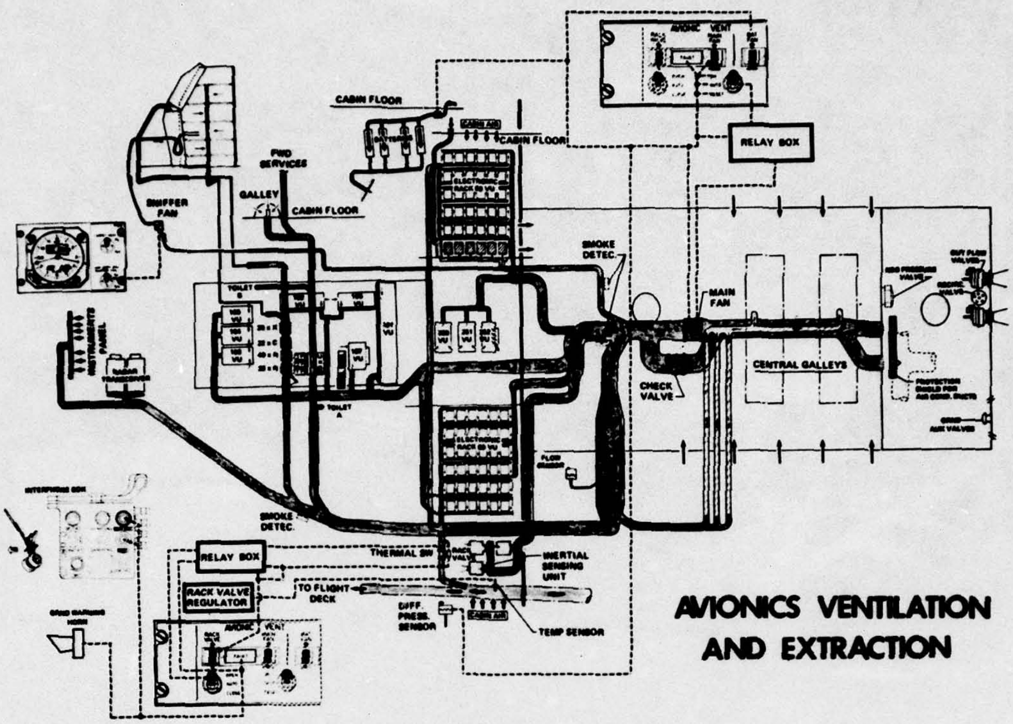

Planche 2 AIRBUS A.300B – Avionics ventilation and extraction

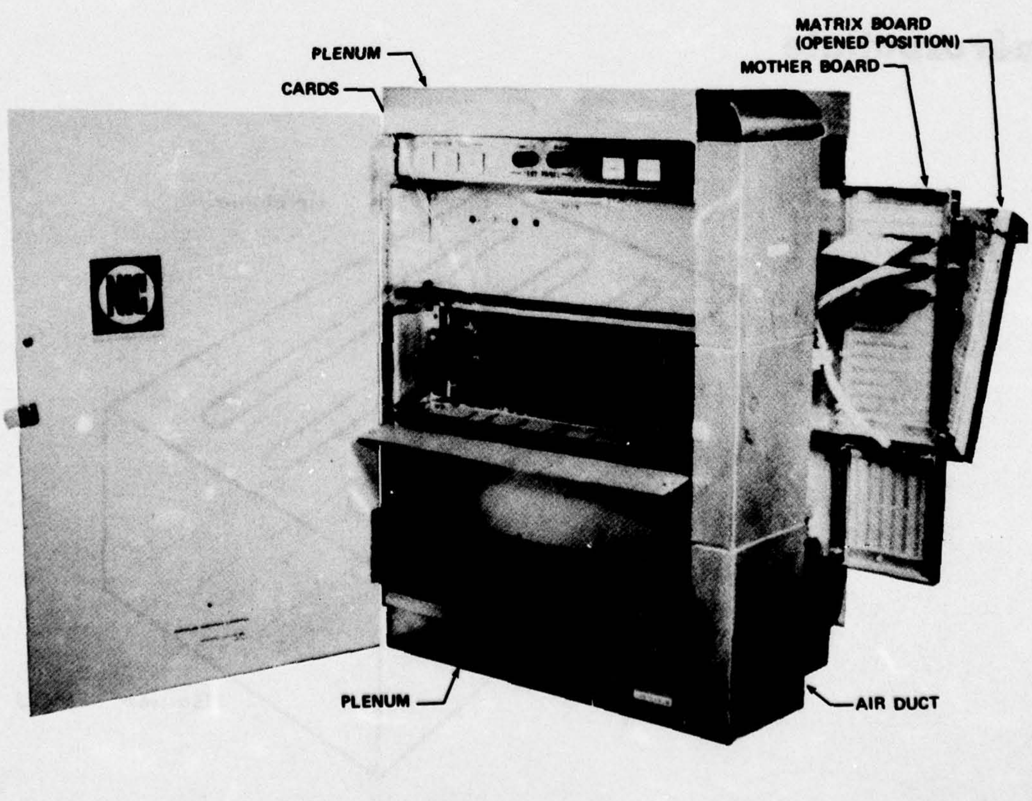

Photo 11 Project NIC
Douglas Aircraft Co

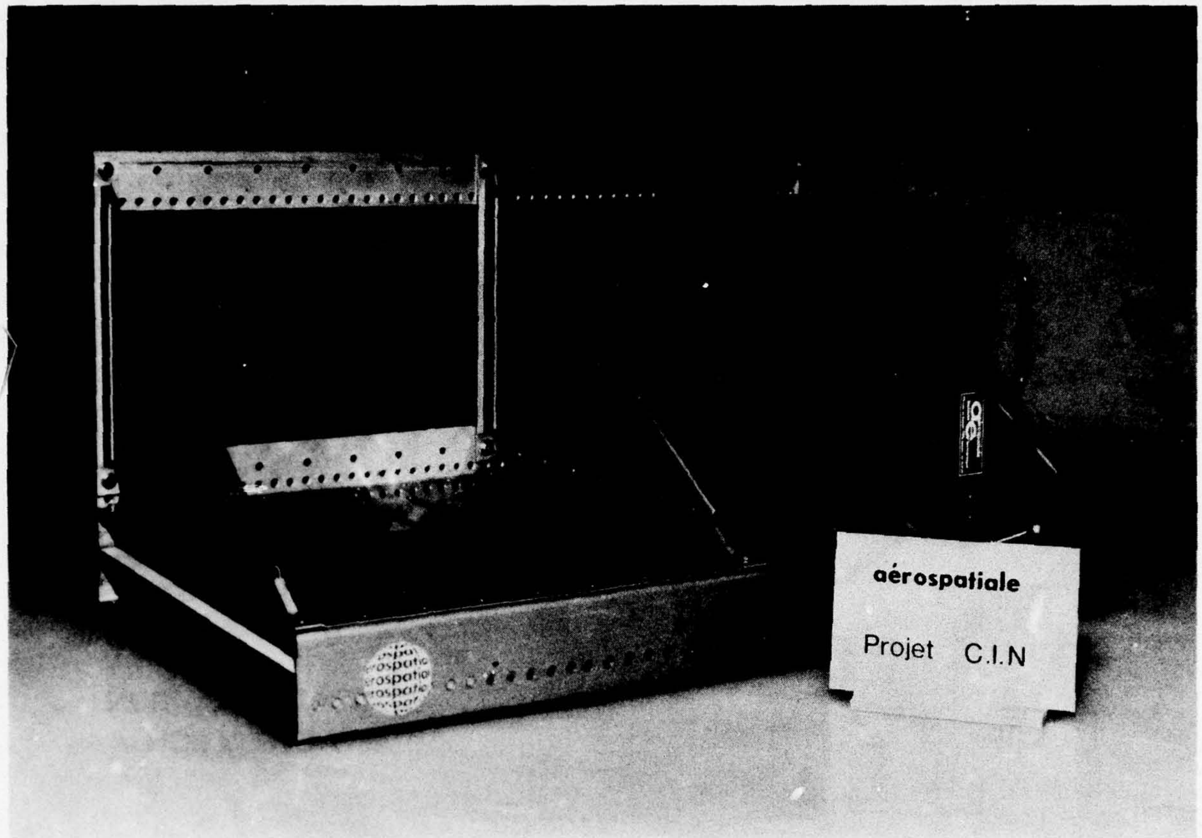

Photo 12 MAQUETTE – Project NIC

aérospatiale

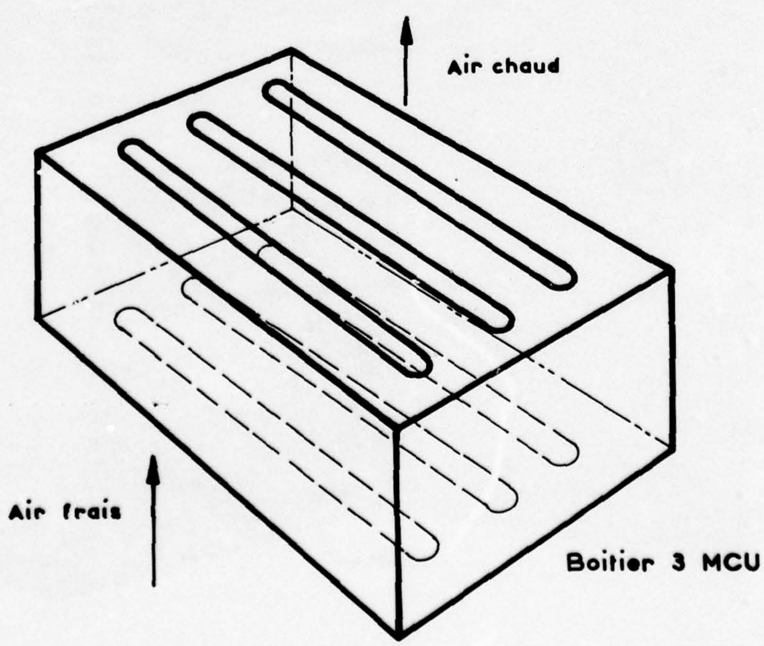

Planche 3 Principe de ventilation

aérospatiale

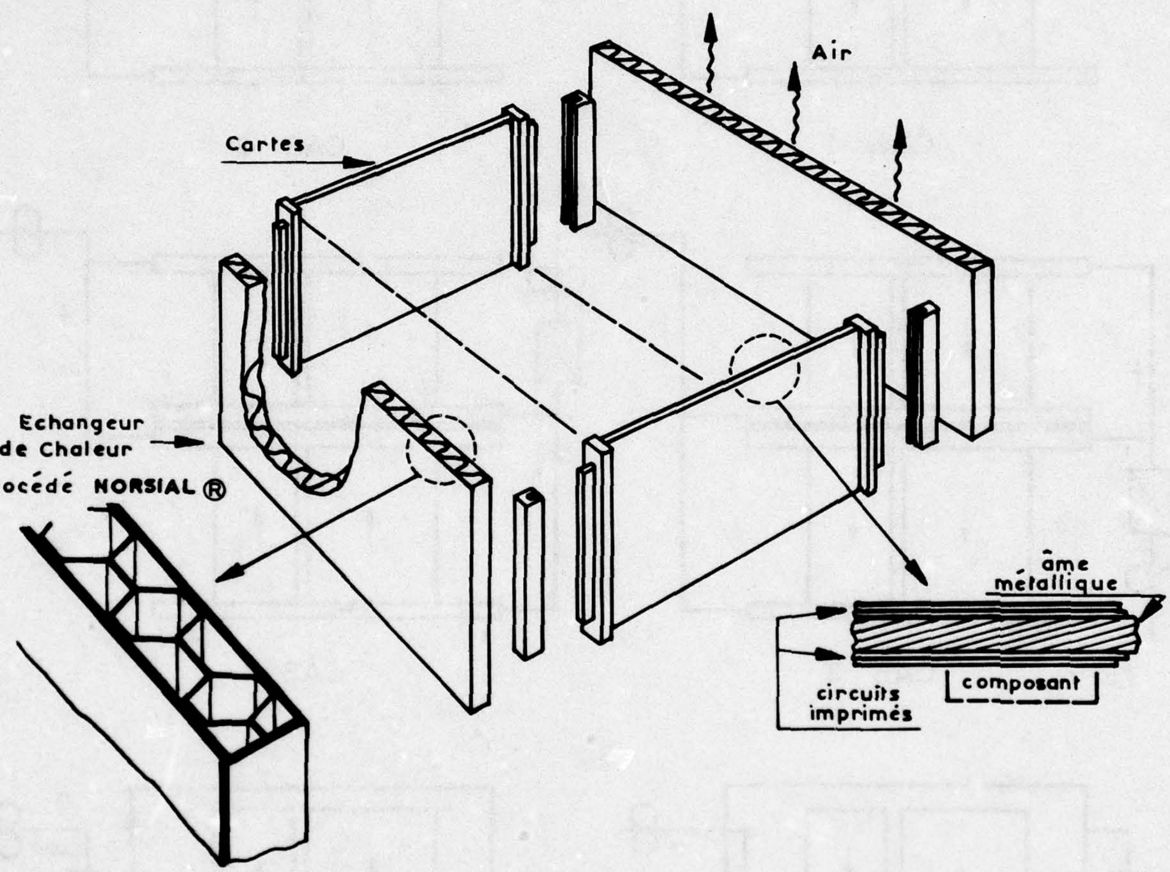

Planche 4 Echangeur de chaleur

aérospatiale

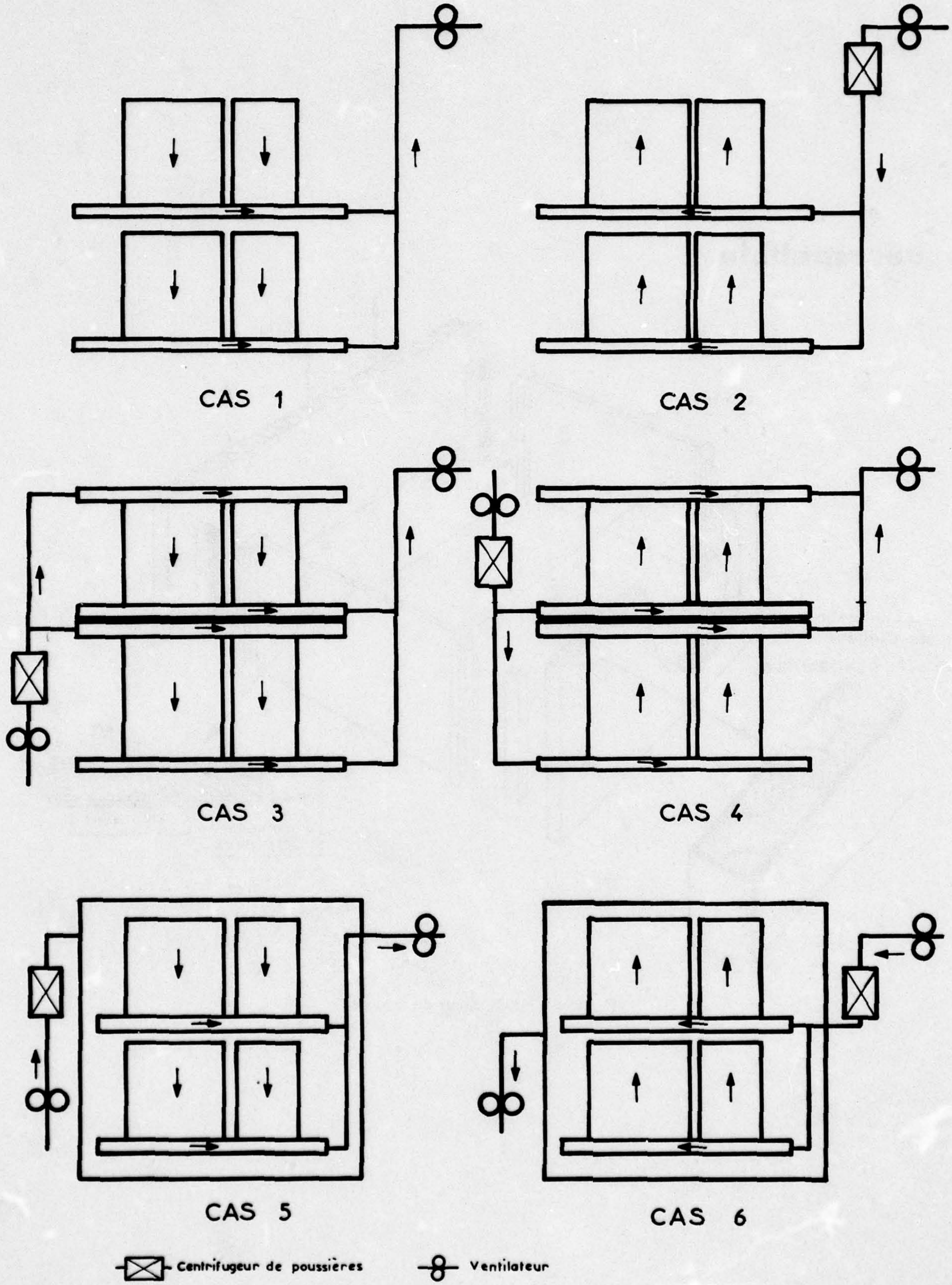

Planche 5 Principes de ventilation

PERFORMANCE ASSESSMENT OF THE CONDITIONING SYSTEM FOR THE AVIONIC EQUIPMENT BAY OF A SMALL HIGH SUBSONIC MILITARY AIRCRAFT.

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England

1. INTRODUCTION

The avionic equipment of a small military aircraft is frequently carried inside a bay in which the thermal environment for the equipment is controlled by an air conditioning system. To analyse the performance and penalties of any conditioning system, the following methods have been developed. They are complementary and provide, when applied to other potential systems, a useful basis for performance comparison.

Energy flow diagram method.

The performance of a system is generally defined by its "efficiency" or "coefficient of performance" which give no indication of the manner in which the input energy varies within the system. This may be overcome by energy flow diagrams which provide a true representation of the energy balance of the system and from which an overall appreciation of the energy variations and losses can be obtained.

Fuel mass penalty method.

The previous method is supplemented by the determination of the overall fuel penalty, incurred in flight, due to system mass, additional drag and power required to operate it. It is shown that to compare several potential systems on the basis of their basic mass only could be highly misleading unless the fuel penalties are also taken into account.

2. DESCRIPTION OF THE CONDITIONING SYSTEM

The type of conditioning plant frequently chosen for providing a controlled thermal environment for the avionic equipment is the turbo-fan system depicted on Fig. 1.

Engine bleed air is cooled by ducting it through a heat exchanger (in which heat is rejected to ram air) and a cold air unit, in which the work done by the air expanding through the turbine is absorbed by a fan inducing ram air through the heat exchanger. The cold bleed air is then mixed with uncooled bleed air in proportions governed by a temperature control valve which ensures that air is supplied to the equipment compartment at the lowest temperature (about 30°C) which would prevent any risk of condensation.

After its passage through the heat exchanger and the fan, ram air is simply discharged overboard without providing any thrust.

3. PERFORMANCE ASSESSMENT OF THE CONDITIONING SYSTEM

3.1 Flight conditions and assumptions

Performance assessment of the conditioning system will be carried out for the following flight conditions:

Altitude	Sea level
Mach number	M = 0.9
Climatic conditions	ISA
Flight duration	45 minutes

The methods for assessing the performance of a system will first be applied to the conventional turbo-fan system in order to demonstrate their validity and usefulness.

They will subsequently be applied to the following systems:-

a) Regenerative system
b) Closed air cycle system
c) Vapour cycle system

which, as will be shown later, exhibit significant improvements over the current turbo-fan system.

For the purpose of comparison between the systems, the following assumptions are made:-

1. The electrical generation system provides a net supply of 8 kW for avionics and the aircraft electrical equipment, as well as any additional electrical power – where applicable – for the conditioning system. The 8 kW supply is entirely converted to heat.

2. Bleed air power is identical on both the conventional turbo-fan and regenerative systems (same engine tapping).

3. The airflow through the avionic bay is the same on all four conditioning systems.

4. The temperature of the fuel used as a heat sink (vapour cycle system) is taken as 50°C.

5. The temperature of the air at inlet of the compartment is taken as 30°C. This will result in virtually no free water in the cooling air at any operating conditions.

6. Heat loads due to kinetic heating are unlikely since the compartment temperature will be about the same as that of the boundary layer.

7. Because of the increased installational complexity of the alternative cooling systems, their basic masses will be greater than that of the turbo-fan system by the following percentages:

   10% for the regenerative and closed air cycle systems;

   50% for the vapour cycle system.

   The increases in basic mass allow, where applicable, for structural refinements (sealing of compartment), provision for thrust recovery, increased generation capacity, increases in mass of the electrical equipment required for operating the conditioning system, additional mass due to the refrigerant (vapour cycle system), etc.

8. The fuel flow penalty due to bleed air and mechanical off-take is the additional engine fuel flow required to maintain constant thrust.

9. The pressure ratio for the closed air cycle is taken as 2.0.

3.2 General remarks on performance assessment

When assessing the performance of an aircraft system, two main objectives must in general be considered:-

a) The power required for its operation and the way in which it varies throughout the system. Such an assessment is best achieved by means of the "Energy flow diagram" method (para. 3.3).

b) The overall fuel mass penalty incurred in flight for carrying and operating the system (para. 3.4).

3.3 Energy flow diagram method. (Ref. 1)

Although the performance of a system may be defined by its "efficiency" or "coefficient of performance" which relate the output power to the input power, no indication is however given of the manner the energy varies within the system or where the losses occur, unless reference is made to the actual cycle calculations.

This shortcoming may be overcome by considering the energy flow diagrams which have the great advantage of providing a pictorial, yet mathematically accurate, representation of the energy balance of the whole system, from which an overall appreciation of the energy variations can be obtained at a glance and the losses easily pin-pointed.

Such a method was applied to the conventional turbo-fan conditioning system of the avionic equipment bay of a small single seat strike aircraft. The resulting diagram is shown on Fig. 2 on which the energies are represented by continuous streams whose width varies proportionally with the energy changes occuring in the system.

The electrical power stream varies due to component losses and is expressed by:-

$$P_{out} = \eta P_{in}$$

It can be seen that 12.5 kW are required from the engine to generate 8 kW for the aircraft avionics (and other electrical equipment), which, for the purpose of the investigation only, are assumed to be entirely converted into heat.

This heat of 8 kW is removed by the conditioning system which is represented by the bleed and ram air streams. The width of each stream is proportional to:

$$Q = mc_p (T - T_\alpha)$$

and it can be seen that the value of Q will be positive or negative depending on whether T is greater or smaller than T_α. Streams representing airflow energy can therefore be assigned a datum line which will indicate whether the air total temperature T is above (Q positive) or below (Q negative) the ambient temperature T_α.

The major part of the air bled from the engine flows through the heat exchanger and the turbine of the cold air unit, at which stage its energy becomes negative (−4.5 kW) since the air total temperature (−6°C) is below the ambient temperature (15°C). The air is then mixed with the uncooled remaining bleed air before entering the avionic equipment bay at a temperature of 30°C, corresponding to an energy of 3.4 kW.

The somewhat complex stream pattern at the CAU turbine on Fig. 2 is explained in detail in Appendix 1.

The reduction in energy of the bleed air, due to heat transfer (53.9 kW) in the heat exchanger and expansion (19.2 kW) through the CAU turbine results in a corresponding increase in energy of the ram air flow which, owing to the forward speed of the aircraft already has an initial energy of 63.9 kW.

If it is assumed that the final ram air energy is entirely dissipated without thrust recovery, as is generally the case, Fig. 2 shows that to generate an electrical supply of 8 kW and provide satisfactory cooling, the total expanded energy reaches 152.8 kW, i.e. 19.1 kW for every net kW of electrical power supplied.

As indicated on Fig. 2, the CAU shaft power is 19.2 kW.

Any improvements in cooling systems should aim, whenever possible, for a substantial reduction of the energy required for cooling purposes as this would lead to a worthwile reduction in fuel mass penalty. Although this particular aspect is dealt with at greater length in paragraphs 3.4 and 4, it is however worth making at this stage a comparison, shown in Table I, of the energies involved between the conventional turbo-fan system and the three improved conditioning plants which will be analysed later. For unchanged design conditions, Table I shows that the lowest power extracted from the engine is 2.17 kW for every kW of electrical power supplied and this is achieved with the vapour cycle system.

It can also be seen that the energy required to provide cooling represents, in each case, an appreciable proportion (greater than 80%) of the total expended energy, except on the vapour cycle system for which the corresponding percentage is 28.

TABLE I. POWERS (kW) INVOLVED AND EXPENDED ENERGIES REQUIRED TO PROVIDE COOLING FOR THE AVIONIC AND ELECTRICAL EQUIPMENT

SEA LEVEL – M = 0.9 – ISA CONDITIONS

	Electr. Gener. System	ALTERNATIVE CONDITIONING SYSTEMS			
		Turbo -fan	Regen- erative	Closed Air Cycle	Vapour Cycle
Mechanical power from engine to generate electrical power	12.52	–	–	25.83	4.83
Bleed air power	–	76.45	76.45	–	–
Ram air drag power	–	63.86	–	63.86	–
Thrust power	–	–	13.31	28.07	–
Energy expended for cooling purposes	–	140.31	63.14	61.62	4.83
Total expended energy for generating electricity and providing cooling	–	152.83	75.66	74.14	17.35
Percentage of the total expended energy attributed to conditioning system	–	91.8%	83.5%	83.1%	27.8%
Total expended energy required for every net kW of electrical power supplied	–	19.10	9.46	9.27	2.17
Energy extracted from engine for every net kW of electrical power supplied	–	11.12	11.12	4.79	2.17

3.4 Overall fuel mass penalty method

The information provided by means of the energy flow diagrams is not sufficient for evaluating the overall effect of the system on the performance of the aircraft or, preferably, for determining the overall fuel mass penalty incurred in flight by that system.

Fitment of a system to an aircraft will result in some increase in fuel required:

a) to carry the system basic mass ΔW_1 by providing additional thrust to match the associated mass drag.

b) to supply power to the system. This is expressed as the fuel flow penalty Δf_s required to maintain constant thrust when power to operate the system is extracted from the engine.

c) to overcome any additional drag ΔD_D which may result from fitment of system to aircraft (increase in profile drag; momentum drag due to ram air induction for cooling purposes; etc.)

d) to carry the resultant quantity of fuel required for items (a), (b) and (c).

The overall fuel mass penalty for a given flight duration under known conditions is derived from reference 2 and given by the general expression:

$$\Delta W_{F_0} = (\Delta W_1 + r\Delta D_D + \frac{r}{c}\Delta f_s)\,(e^{\frac{c}{r}t} - 1)$$

Table II lists the values of the relevant terms appearing in the above expression for a flight duration of 45 minutes and for the four conditioning systems under consideration.

In spite of its lower basic mass (87.5 lb.), the conventional turbo-fan system will exhibit a comparatively large fuel mass penalty (192.6 lb.) due mainly to ram air momentum drag and, to a lesser extent, the additional fuel (42.0 lb.) required to maintain constant thrust when air is bled from the engine compressor.

With reference to the expression for the overall fuel mass penalty ΔW_{F_O}, it can be seen that, although the value of the basic mass (87.5 lb) of the turbo-fan cooling system is of the same order as that for the ram air momentum drag (93.6 lbf), the fuel mass penalty (110.8 lb) of the latter is appreciably greater, roughly by a factor r, than the fuel mass penalty (39.8 lb) due to the basic system mass.

This highlights in particular the relative importance between the fuel mass penalty due to additional drag and that due to basic mass. Any advantage gained in selecting the lightest system could be lost if its installation resulted in some additional drag, especially in cruise when high values of the lift/drag ratio r would normally prevail.

It follows that, when analysing the performance of a system, its overall effect should be considered and not only whether it exhibits the lowest basic mass. This is clearly seen from Table II which also shows that the total fuel mass penalty for every net kW of electrical power supplied drops from 36.3 lb. for the conventional turbo-fan system to about 20 lb. for both the regenerative and fuel cooled vapour cycle systems, and 24.2 lb. for the closed air cycle system.

4. IMPROVEMENT POSSIBILITIES

4.1 Reduction of the overall fuel mass penalty

Apart from any reduction in the basic mass of a system, any system improvement should also aim for a reduction of the overall fuel mass penalty, made up of the items (a), (b), (c) and (d) listed in paragraph 3.4.

Because items (b) and (c) are fuel penalties due basically to drag or power extraction effects (and not mass), an appreciable saving in fuel could be achieved if the power required to operate the system and overcome any additional drag could be reduced significantly (ref. 2).

This may be achieved at the expense of an increase in basic system mass, but if the drag or power savings outweigh the increased basic mass, resulting in reduced total mass of equipment and fuel, then an improvement has been made. The extent of this improvement will depend on mission time. The drag or power savings decrease for shorter missions and there is a "break-even" mission time below which fitment of an improved but heavier system would not be advantageous in terms of total mass.

In the following paragraphs, an attempt will be made to analyse some of the more promising possibilities for improving the conventional turbo-fan system and also examine the potentials of the regenerative system (which is self-cooled), the ram air cooled air closed cycle system and the fuel cooled vapour cycle system (Fig. 3). The bootstrap system is unlikely to be a suitable contender – in spite of the lower bleed pressures required – because of the comparatively large quantities of heat to be dissipated which may exceed those of the conventional turbo-fan system.

4.2 Turbo-fan system

Because of its basic design characteristic, the current turbo-fan system (Figs. 1 and 2) will always be associated with high fuel mass penalties due to momentum drag of the ram air ducted through the heat exchanger and the fan. In the case under consideration, this penalty reaches 192.6 lb.

Reduction in fuel mass penalties could be achieved by adopting a two-stage tapping device (which would ensure a better utilisation of bleed air energy) and by discharging ram air through aft facing nozzles, thus recovering some thrust.

Because of the losses incurred throughout the system, total thrust recovery is most unlikely, in spite of the increase in temperature of ram air through the heat exchanger. In the flight case under consideration, the thrust recovered would be about 65 lbf, resulting in a fuel mass saving of 77 lb. This is 34 lb. short of 111 lb. corresponding to full thrust recovery.

Although the fuel saved would be partly offset by the fuel penalty incurred by the additional mass of the nozzle and the associated ducting, such a modified turbo-fan system would exhibit a total mass which could compare favourably with that of the other systems considered in the following paragraphs.

4.3 Regenerative system

Satisfactory operation of the regenerative system (Figs 3 and 4) depends on adequate sealing of the equipment bay to ensure that leaks are negligible and the heat exchanger flow ratio remains near unity. Efficient utilisation of bleed air could again be achieved by means of a two-stage tapping device and thrust recovery may further help in reducing the total expended energy. As can be seen from Fig. 4, this is however reduced by only 13.3 kW (Appendix 2).

4.4 Closed air cycle system

On the closed air cycle system (Figs. 3 and 5), bleed air power from the engine is replaced by mechanical power and this results in a significant saving in fuel since 25.8 kW (Table I) only are extracted from the engine for its operation compared with the bleed air power of 76.5 kW required for the conventional turbo-fan system. (Appendix 3).

Cooling is provided by ram air, the adverse effect of which is however reduced by nearly half by thrust recovery (Appendix 2). The quantity of heat transferred to ram air amounts to 21.1 kW which represents only 39% of the heat rejected through the heat exchanger of the conventional turbo-fan system. But no reduction in the cooling flow (and the associated drag penalty) can be considered unless the flow ratio can also be altered.

TABLE II. FUEL PENALTY

SEA LEVEL – M = 0.9 – ISA CONDITIONS – 45 MINUTES

	Electr. Gener. Syst.	ALTERNATIVE CONDITIONING SYSTEMS			
		Turbo-Fan	Regen-erative	Closed Air Cycle	Vapour Cycle
Basic mass (lb)	200.00	87.50	96.25	96.25	131.25
Air flow (lb/sec)	–	0.505	0.505	0.505	0.505
Ram air or cooling flow (lb/sec)	–	3.00	0.454	3.00	–
Thrust (lbf)	–	–	17.54	62.08	–
Ram air drag (lbf)	–	93.6	–	93.6	–
Fuel penalty (lb) due to:					
Basic mass	91.00	39.81	43.79	43.79	59.72
Bleed air	–	41.98	41.98	–	–
Mech. off-take	6.88	–	–	14.18	2.65
Ram air drag	–	110.76	–	110.76	–
Thrust recovery	–	–	–20.74	–73.43	–
Total penalty	97.88	192.55	65.03	95.30	62.37
Total mass (lb)	297.88	280.05	161.28	191.55	193.62
Mass reduction (lb) (1)	–	0	118.77	88.50	86.43
Additional payload (lb) (1)	–	0	81.63	60.83	59.40
Total fuel mass penalty for every net kW of electrical power supplied (lb/kW) (2)	–	36.30	20.36	24.15	20.03
Break-even time (min) (1)	–	–	3.7	13.5	23.8

(1) Based on turbo-fan system

(2) Based on: (97.88 + total fuel penalty of relevant conditioning system) /8.

4.5 Vapour cycle system

In spite of its increased mass, due partly to that of the refrigerant, the vapour cycle system (Figs. 3 and 6) exhibits a lower overall fuel mass penalty due to the more efficient use of mechanical power – instead of engine bleed air power – for its operation for which 4.8 kW only are required from the total power of 17.4 kW extracted from the engine. As in the regenerative system, it does not depend on ram air cooling, heat being rejected to the aircraft fuel at an assumed temperature of 50°C (Appendix 4).

The amount of heat (10.6 kW) rejected to the fuel represents only about a fifth of the heat transferred to ram air in the conventional turbo-fan system.

4.6 Practical considerations

The improvements in conditioning are likely to be achieved at the expense of a system mass increase – hence increase in fuel mass penalty – due to some increased installational complexity. However, it will be noted, from Table II, that the resulting fuel mass penalty would be small compared with the fuel penalty which would otherwise be incurred by excessive drag and, as a result, additional payload may even be contemplated as can be seen from Table II (81.6 lb, 60.8 lb, and 59.4 lb. for the regenerative, the closed air cycle and vapour cycle systems respectively). Based on this table, Fig. 7 illustrates the significant savings in fuel achieved with the alternative conditioning systems, in spite of their increased basic masses.

Of the four conditioning systems considered, the regenerative system is the only one which is self-cooled since bleed air is later also used for cooling. Although ram air is generally used as a heat sink, the aircraft fuel may be considered as a suitable alternative, especially as it is not associated with any drag penalty. However, the cooling potential of the fuel may be limited and care must be exercised to ensure that it remains adequate throughout the mission. The use of surface heat exchanger may also provide cooling free from drag effects (unless such a cooler is mounted inside the engine intake) as heat would be rejected through the aircraft skin to the adjacent boundary layer. For a system flow of 0.45 lb/sec, about 53 kW could be dissipated over an area of 12 ft^2. at M = 0.9 at sea level (ref. 3).

Tapping air from the engine compressor has become an undesirable, yet often unavoidable feature of aircraft systems, especially when large amounts of bleed air are required. Of the four cooling systems considered in this investigation, the vapour and closed air cycle systems do not depend, for their operation, on bleed air, but on mechanical off-take from the engine, which is more efficient, as can be seen from the corresponding fuel penalties given on Table II.

A comparison of the total mass of the systems at the beginning of a sea level cruise at M = 0.9 under ISA conditions is shown on Fig. 7.

As indicated on Table II, fitment of the improved – but heavier– alternative conditioning systems would prove advantageous provided the duration of the mission exceeds the "break-even" time which is 3.7 min., 13.5 min. and 23.8 min. for the regenerative, the closed air and vapour cycle systems respectively (Appendix 5).

Should thrust recovery by means of rear facing nozzles prove difficult to achieve, partial recovery could be obtained by ram air modulation.

5. CONCLUSIONS

The type of conditioning plant frequently chosen for cooling the avionic equipment of a small high subsonic military aircraft is the turbo-fan system in which a total of 19 kW is expended for every net kW of electrical power supplied to the avionics during a sea level cruise at Mach 0.9 under ISA conditions, the corresponding total fuel penalty incurred being 36 lb/kW.

Reduction in the required cooling power may be achieved by replacement of the conventional turbo-fan system by improved alternative conditioning plants such as the self-cooled regenerative system, the ram air cooled closed air cycle system and the fuel cooled vapour cycle system. In the three cases, the total power expended for every net kW of electrical power supplied would drop to less than half the turbo-fan value and the total fuel penalty to less than 2/3 of the corresponding value.

The improvements would be achieved at the cost of some additional installational complexity and, hence some unavoidable increase in the basic mass of the alternative systems. These improvements are partly due to thrust recovery (regenerative and closed air cycle systems), to mechanical off-take power (closed air and vapour cycle systems) instead of bleed air power, and to fuel cooling (vapour cycle system).

Power extraction from the engine would be more efficiently achieved by means of a mechanical drive than bleeding air from the engine compressor. Not only would the adverse effects on engine performance be avoided – especially when large amounts of bleed air are required – but the power extracted mechanically from the engine would be appreciably reduced.

Further improvements could be achieved by suitable selection in flight of the compressor tapping for better utilisation of bleed air or by making a greater use of fuel for cooling purposes, thus reducing or even eliminating completely ram air cooling and its associated momentum drag. Should the cooling potential of the fuel prove however inadequate, consideration could also be given, where applicable, to the use of surface heat exchangers through which heat would be rejected to the adjacent boundary layer.

6. NOTATION

c	Specific fuel consumption	lb/hr lbf
c_p	Specific heat	CHU/lb °C
m	Airflow	lb/sec.
M	Mach number	–
P_{in}	Input power	kW
P_{out}	Output power	kW
Q	Heat flow	CHU/sec.
r	Lift/drag ratio	–
t	Time	hr.
T	Temperature	°K
T_α	Ambient temperature	°K
ΔD_D	Additional drag of system (other than mass drag)	lbf
Δf_s	Fuel flow penalty due to operation of system	lb/hr
ΔW_{F_O}	Fuel mass at beginning of cruise required for taking the system mass ΔW_1 over the distance covered in time t, for operating the system and for overcoming any additional drag ΔD_D	lb

ΔW_1	System basic mass	lb
η	Efficiency	–

7. REFERENCES

1. Le Claire, R. Energy Flow Diagrams. Aeronautical Journal, August 1972.

2. Le Claire, R. Evaluation of the overall fuel mass penalty of an aircraft system. Aeronautical Journal, May and July 1975.

3. Le Claire, R. Surface Heat Exchangers, Aeronautical Quarterly, February 1976.

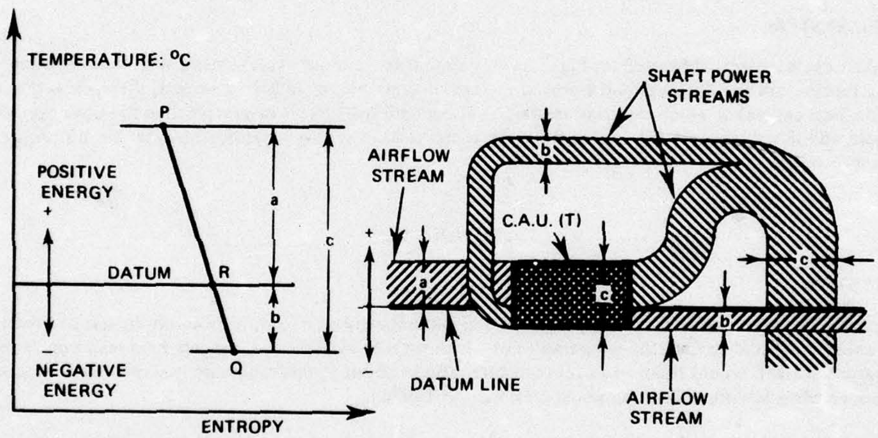
ACKNOWLEDGEMENT

The author would like to express his thanks to Hawker Siddeley Aviation Ltd., for permission to publish this paper.

APPENDIX 1

ENERGY PATTERN AT THE CAU TURBINE

Consider the energy pattern at the CAU turbine shown on Figs. 2 and 4. The expansion of air through the CAU turbine, which is accompanied by a drop in temperature from P to Q, results in work required to drive the fan. On the temperature-entropy



diagram, this is represented by the curve PRQ which crosses the datum line for T= 15°C at R.

While the energy stream for air varies from a positive value 'a' (above the datum line) at turbine inlet to a negative value 'b' (below the datum line) at turbine outlet, the overall work done by the turbine is represented by the quantity 'c' which, for clarity and convenience, is shown by two branches of the common stream on the energy flow diagram.

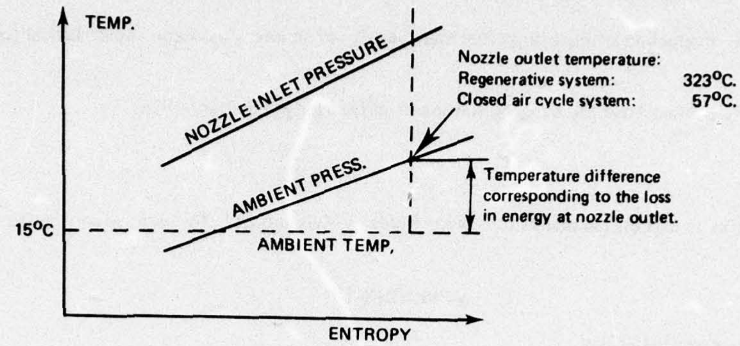
APPENDIX 2

REGENERATIVE AND CLOSED AIR CYCLE SYSTEMS

On the regenerative and closed air cycle systems (Figs. 4 and 5), the energy recovered due to thrust is 13.3 kW and 28.1 kW respectively. There is no possibility of improving still further the energy recovered since, as can be seen from the temperature-entropy diagram, the nozzle outlet temperature is 323°C and 57°C for the regenerative and closed air cycle systems respectively, and therefore well above the ambient temperature.

This will result in a loss in energy of 71.0 kW on the regenerative system which, combined with the losses of 4.5 kW and 0.2 kW incurred by the electrical generation system and the CAU respectively, gives a total expended power of 75.7 kW (Fig. 4).

In the case of the closed air cycle system, the corresponding loss at nozzle outlet is 56.9 kW, which, added to the remaining expended power of 17.2 kW incurred by the electrical generation system, results in total expended power of 74.1 kW (Fig. 5).



APPENDIX 3

CLOSED AIR CYCLE SYSTEM

The closed air cycle system, illustrated on Fig. 3, is very similar to a vapour cycle system, with the throttling valve being replaced with a turbine and the working fluid being air instead of a refrigerant. In such a system, the cycle is that of a heat pump (as opposed to a heat engine) in which the compressor power demand will always be greater than the power provided by the turbine and, as a result, additional power is required to drive the compressor. Cooling is provided by ram air, but drag effects are partly offset by thrust recovery.

APPENDIX 4

VAPOUR CYCLE SYSTEM

The analysis of the vapour cycle system, represented diagrammatically on Fig. 3, is based on the use of Freon 21 as refrigerant. Heat is entirely rejected to fuel, the temperature of which is taken as 50°C. (Although heat rejection to the boundary layer air at a temperature of 62°C would result in a cycle pressure ratio of about 5, this ratio may however reach unacceptable values under tropical conditions when boundary layer air could attain about 100°C).

The air circulating fan and the compressor are assumed to be electrically driven, the required additional mass of the electrical generation system to provide the extra power being chargeable to the cooling system. The power required by the fan is estimated at 0.3 kW and, for simplicity, it is assumed that no losses are incurred, the effect on the energy flow diagram (Fig. 6) being insignificant.

The energies of the air cycle are based on an ambient temperature of 15°C.

The energies of the vapour cycle are based on the datum used for the Freon pressure-enthalpy diagram. The corresponding magnitude of these energies is not relevant to the investigation since the refrigerant is merely a means for transferring heat and, therefore, only the difference is of interest. It can be seen from Fig. 6 that the heat rejected to the fuel by the conditioning system only would be 10.6 kW.

APPENDIX 5

BREAK-EVEN TIME

The "break-even" time is defined as the time in flight below which replacement of a system by an improved, but heavier alternative would not prove advantageous in terms of overall mass (Ref. 2). In the case of the vapour cycle system, the "break-even" time was found to be 23.8 minutes, which is just over half the actual flight duration.

When the "break-even" time represents a substantial proportion of the flight, the question may arise whether the switch from one system to another would still be worthwhile – in spite of the resultant increase in payload – if flights under the stated conditions are rather the exception than the rule and their duration usually much shorter.

Thus the choice of system will depend on the expected usage of the aircraft, as given in its specification.

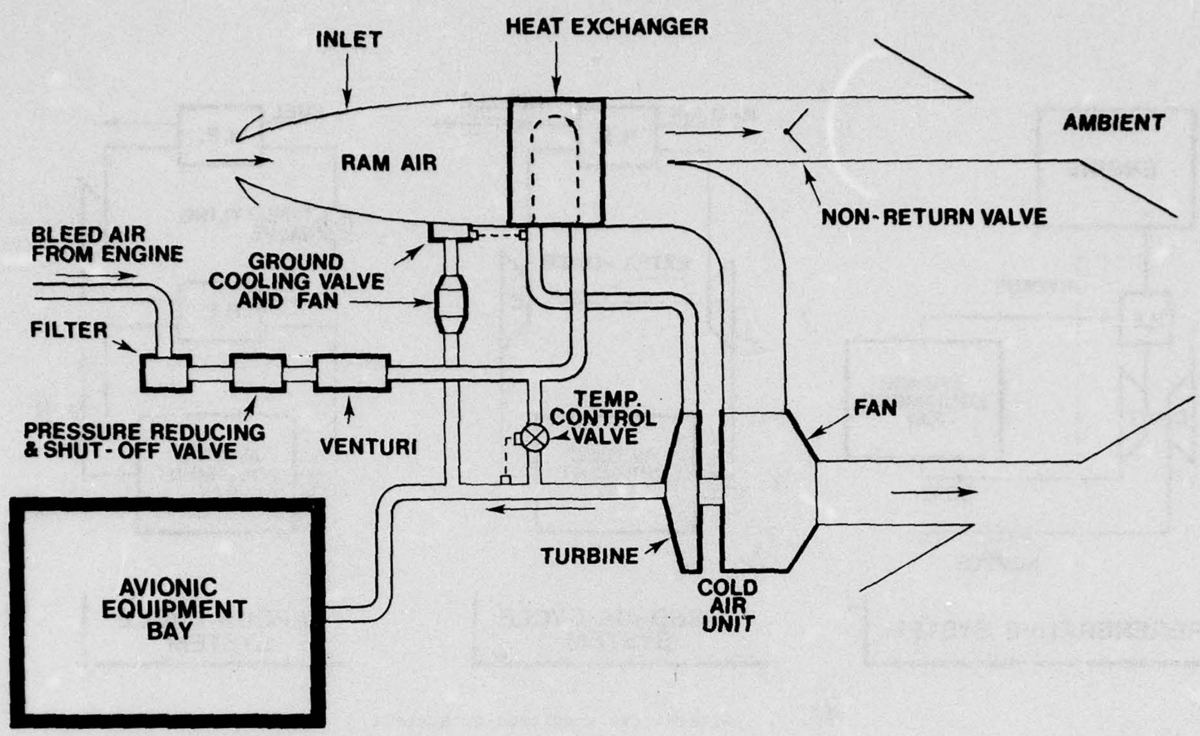


Fig. 1. Conventional turbo-fan conditioning system

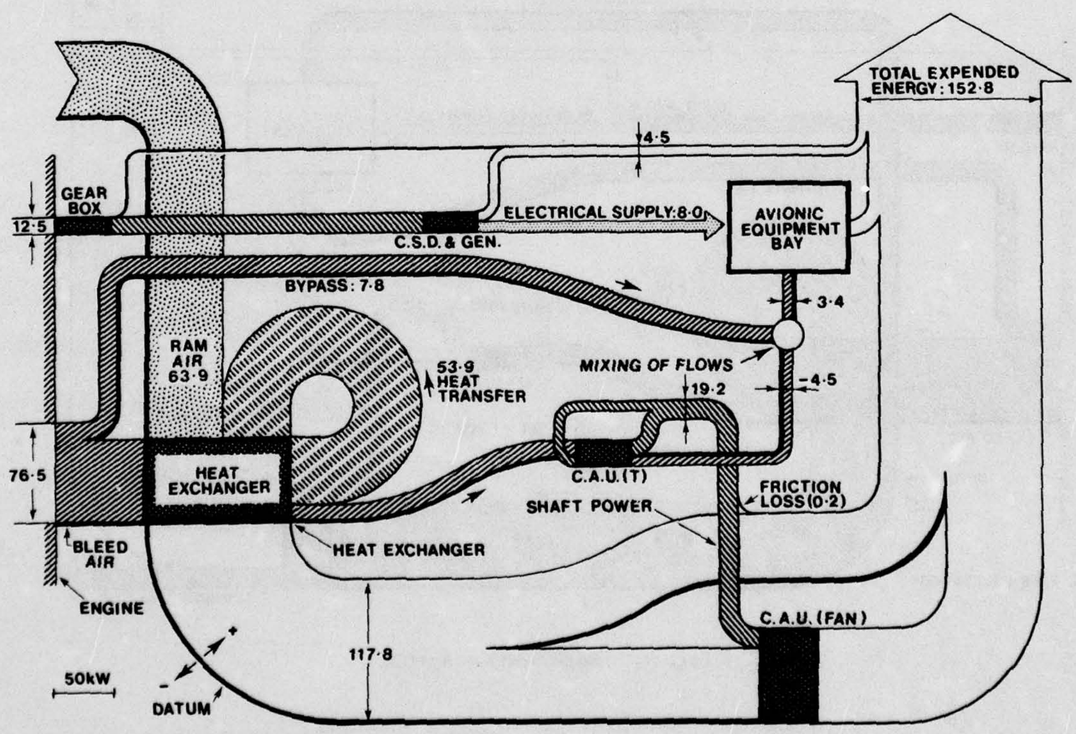


Fig. 2. Turbo-fan system

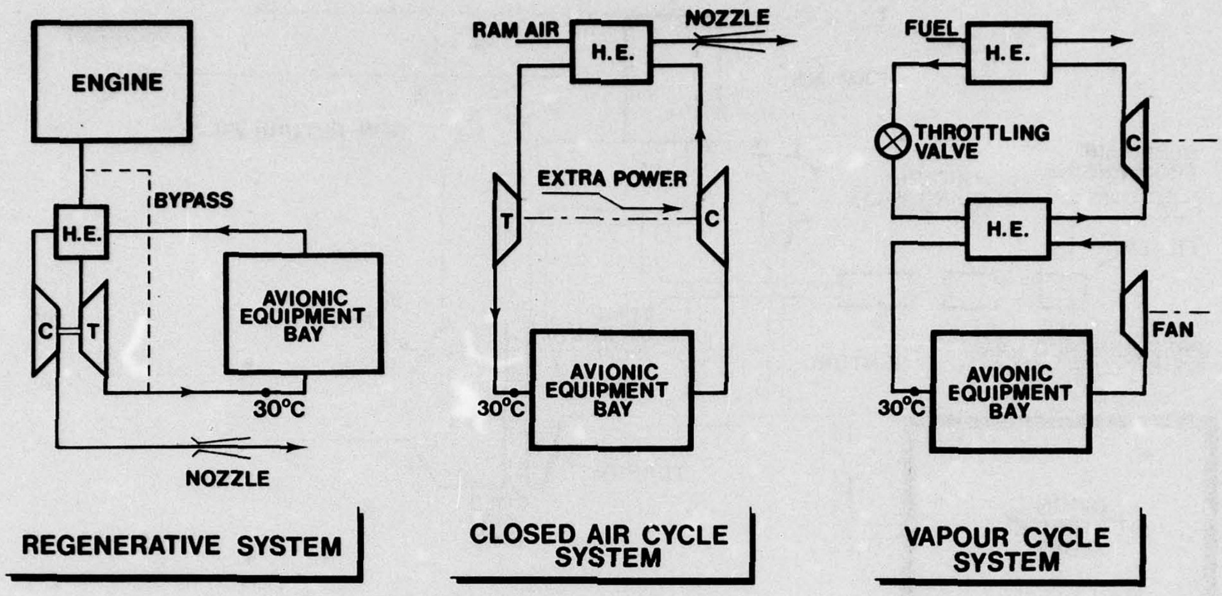


Fig. 3. Alternative conditioning systems

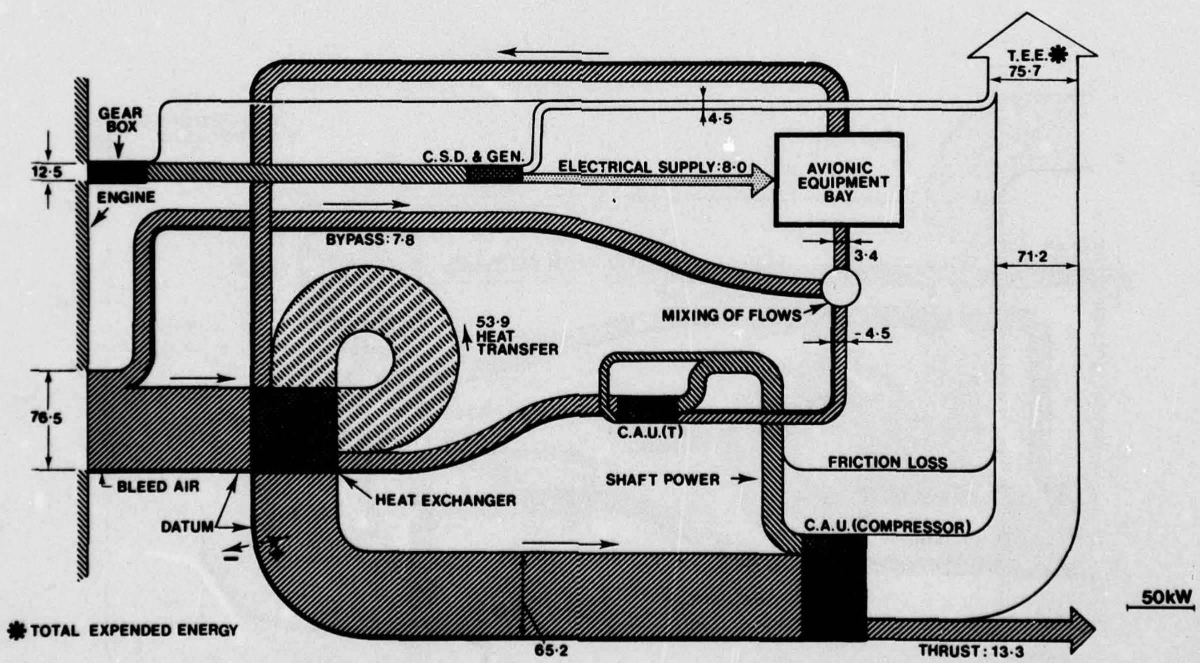


Fig. 4. Regenerative system

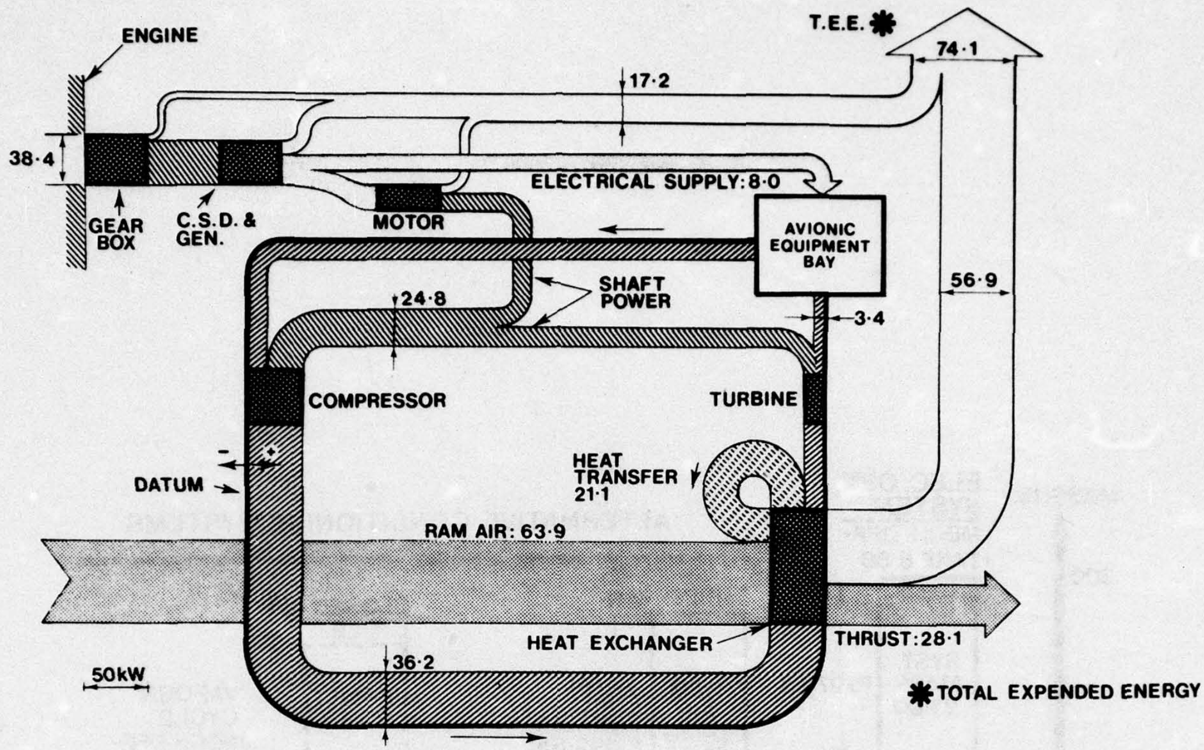


Fig. 5. Closed air cycle

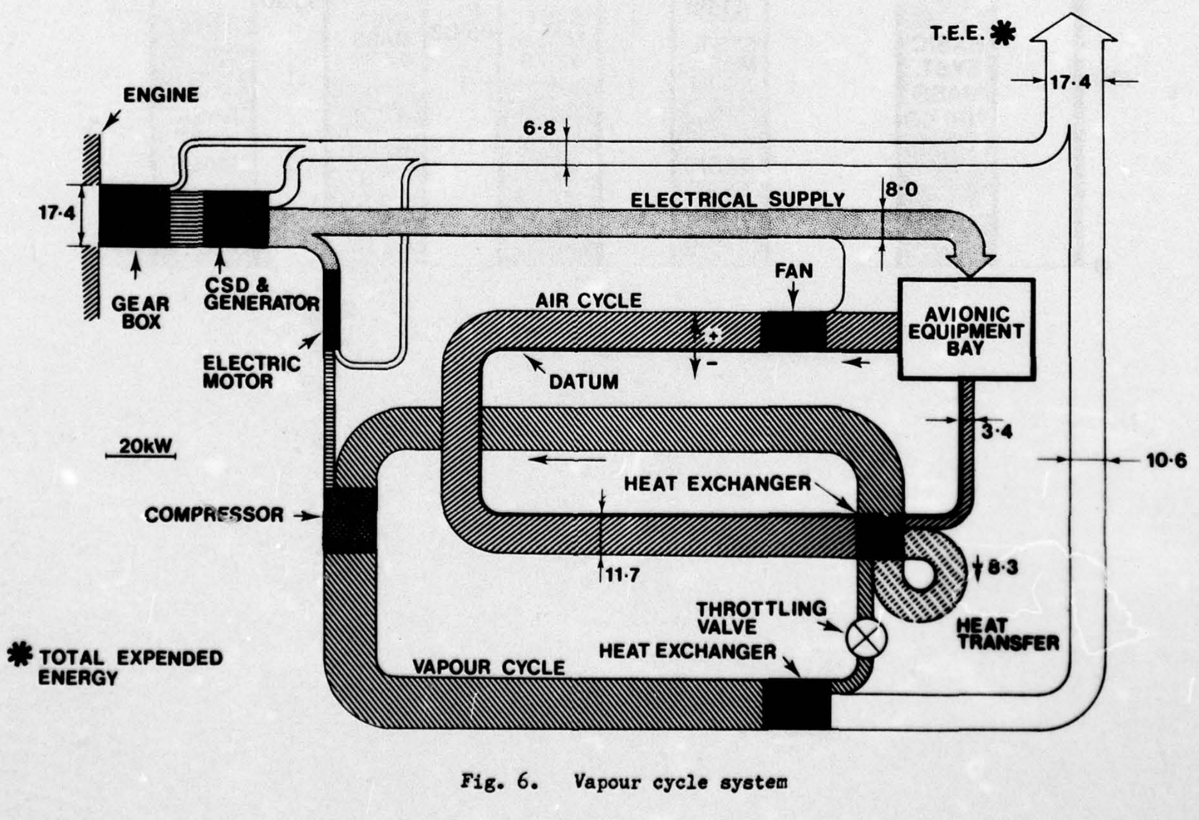


Fig. 6. Vapour cycle system

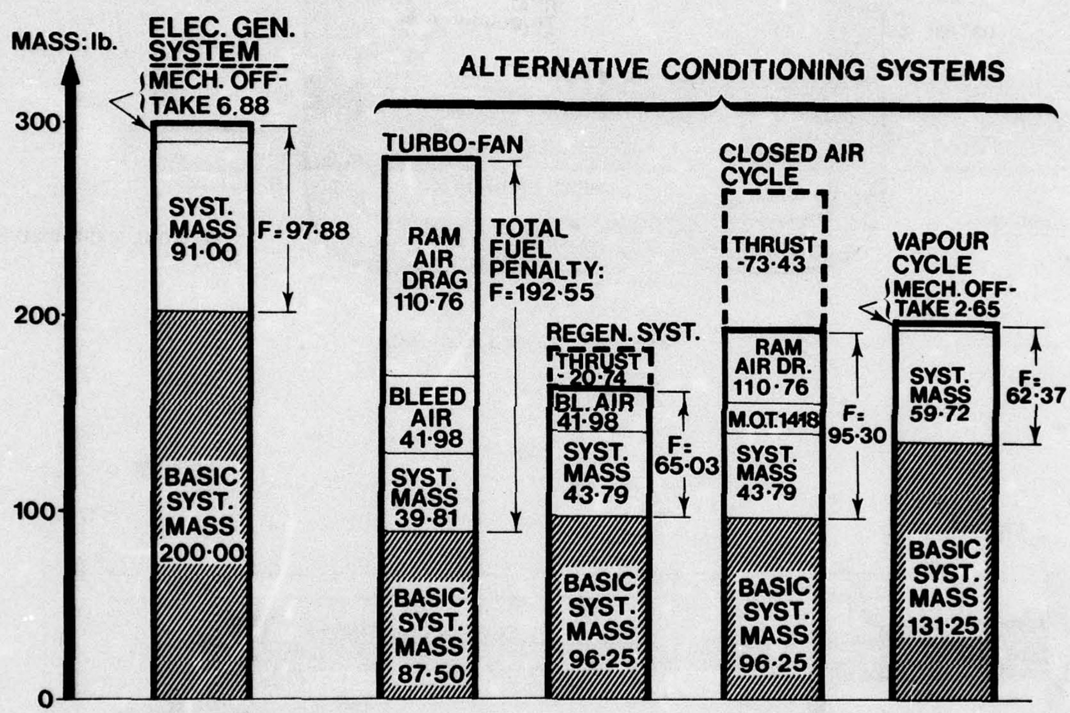


Fig. 7. Total mass

DISCUSSION

G Borgonovo:

The author analysed the system condition using performance with respect to fuel penalty and total expanded energy. The request is to know if any attempt has been made to assess the system performance with respect to its overall cost.

R H Le Claire:

The purpose of the investigation was to develop two complementary techniques for assessing the performance and penalties of environmental control systems. No attempt was therefore made at this stage to consider the overall cost involved, which would of course become necessary should a specific application be envisaged.

D H Mehrtens:

The Energy analysis has been based on the presumption that the cooling air supply is at the relatively high temperature of 30°C. What would be the results when applied to supplying cold air at, say 10°C, which the equipment designer might find more useful?

R H Le Claire:

Two main aspects have to be considered:

a A reduction in bay inlet temperature from 30°C to 10°C would result in a reduction - which could be significant - of the required cooling air flows. This would reduce some of the energies and penalties involved.

b A temperature of 30°C was chosen because this would avoid the risk of condensation in the worst cases of ambient moisture content likely to be encountered. If the inlet temperature is reduced, a water extractor would be necessary, which would extract most of the condensate. The remaining water could however still cause trouble depending on the type of equipment being cooled.

THE EFFECT OF AVIONICS SYSTEM CHARACTERISTICS ON FIGHTER AIRCRAFT SIZE, COOLING, AND ELECTRICAL POWER SUBSYSTEMS

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SUMMARY

The effect of avionic systems electrical power and cooling requirements on overall aircraft size and life-cycle cost are considered. Power and cooling requirements typical of advanced fighter aircraft are first defined; their direct weight and cost influences are then estimated, and a simplified parametric analysis is used to determine the compounding effects on the engine and airframe characteristics brought about by the aircraft "growth curve". It is shown that overall weight and cost effects are significant, particularly when considered in the context of total program life-cycle cost. It is found that technology developments should emphasize system weight reductions.

1. INTRODUCTION

Avionics systems have, over the past several decades, made up a growing proportion of total aircraft life-cycle costs, increasing from an almost negligible percentage in early World War II to typically 25% today. This trend promises to continue.

In addition to their direct effects on weight and cost, avionics systems also have an indirect effect on the aircraft through their influence on other subsystem requirements, particularly on the electrical power (EP) and environmental control (EC) subsystems. This paper specifically addresses the overall influence of these latter subsystems on the total aircraft.

Considering the relatively small direct influence on life-cycle cost that is typical of EP and EC subsystems, as shown in Figure 1, it might at first be supposed that improvements in their characteristics would have little influence on overall aircraft cost and performance. When considering high-performance fighter aircraft, however, this turns out not to be the case, as shown in the following pages.

2. SYSTEM REQUIREMENTS

A significant portion of the net impact of any subsystem on overall aircraft performance comes from the influence of the aircraft "growth curve". As illustrated in Figure 2, if performance is to be maintained when dry weight is added to the aircraft, total aircraft weight must be increased by an amount which is often much larger than the subsystem weight increment. The extra weight comes from the additional thrust, wing area, and fuel necessary to compensate for the added dry weight. This "growth factor" is a strong function of aircraft size and performance level, with smaller, higher-performance aircraft typically being the most sensitive. The growth factor also depends on the number of performance parameters (e.g., range, payload, maneuverability) which are held fixed in sizing the aircraft. On a preliminary design level, the practice is usually to consider all performance objectives to be fixed so that the most direct comparison possible can be made between competing concepts or technologies. This convention is adopted throughout the remainder of this paper.

In order that the overall aircraft impact of any change can be defined, it is first necessary to define component sensitivities. Typical EP and EC system overall capacities are shown in Figure 3, as related to the actual peak avionics power dissipation. As shown, the actual power supplied by the EP system to the avionics is often considerably less than the peak power capacity designed into the generating system to allow for growth capability or to provide redundancy. On several recent General Dynamics aircraft this difference has amounted to a factor of around 5 - the value adopted here. (Other values are of course possible, depending on the circumstances, and can be evaluated by the methodology described in the Appendix.)

Cooling requirements are determined both by the avionics load and by other factors, including solar radiation and aerodynamic heating. A value of 5 kW above the avionics level is assumed for the purposes of analysis. (This value can be considerably greater on aircraft capable of speeds in excess of Mach 2.2 to 2.5, the upper limit of speed considered in the present paper.)

System weight levels for given power requirements have shown a significant downward trend with time, as illustrated in Figure 4 for EP generating systems. Modern systems generally follow the weight and cost trends shown in Figures 5 and 6 for EP and EC systems, respectively, although there is significant scatter in the data. In the following analyses "new" systems are assumed at the nominal weight and cost levels shown, with costs following a 90% learning curve.

In addition to direct weight effects, avionics power extraction also degrades engine performance, both from the direct power extraction and from air-conditioning requirements, which are usually met by refrigeration systems employing high-pressure engine bleed air.

The curves of Figure 7 were derived for a specific advanced, afterburning-turbofan engine cycle and include both bleed and power-extraction effects. Of the two, engine bleed generally causes the most severe penalties. In general, the overall penalties are not large in absolute terms, but can amount to several percent for high-power-extraction levels. Penalties are naturally the greatest for small engines and, at least for the particular engine cycle being considered, are considerably more severe at supersonic speeds. (Other cycles can be expected to have somewhat different characteristics.)

For most aircraft, peak power extraction levels occur for only a small proportion of the total flight time. For this reason, the data presented in the remainder of this paper assumes only a fixed, nominal, engine-power-extraction level (except as noted) that does not change with the design system maximum capacity.

It is interesting to note that in relative terms the complete power-extraction process is highly inefficient. If the propulsion system energy input to the aircraft is assumed to be proportional to the thrust times the flight velocity, the aircraft powered by the engine of Figure 7 will lose approximately 52 kW per kW of power extracted at subsonic speeds, and over 200 kW per kW of power extracted at the Mach 1.6 point. Engine bleed accounts for the majority of this loss, amounting to roughly 92% of the total at the higher velocity.

3. PARAMETRIC AIRCRAFT SIZING RESULTS

Completely resizing an aircraft for different subsystem characteristics can be a time-consuming process when done rigorously; however, it has been found that highly simplified scaling relations, as outlined in the Appendix, can provide reasonably accurate estimates of major trends. These techniques were employed to derive the results presented on the following pages. They can easily be extended to other cases of interest by employing the equations of the Appendix. Aircraft characteristics assumed are those of a typical advanced-technology supersonic fighter in the Mach 2 class.

Figure 8 provides a breakout of the way in which overall aircraft gross weight typically changes with the required avionic power (only avionics power levels are changed, avionic weights are held fixed). Weight variations of the EC and EP systems are as shown in Figures 5 and 6. It should be noted that the horizontal axis in both figures is the actual design peak power dissipation in kW (the EP system maximum capacity in kVA is assumed to be 5 times this value, as shown in Figure 3).

It is apparent from Figure 8 that EP and EC influences on aircraft gross weight can be significant if high power levels are required, and that airframe, engine, and fuel growth is several times the incremental weight of the systems themselves.

Even more importantly, if the weight growth is converted into cost terms over the fleet life cycle, the true magnitude of the effects being considered can be discerned. As shown in Figure 9, the total-force life-cycle cost for the specific high-performance air-to-air fighter assumed can be expected to be in the range of $12 billion (FY 1975 dollars). The range of power extraction from 0 to 24 kW causes a corresponding total-force life-cycle cost difference of over $1 billion. Fuel cost changes alone are of the same order of magnitude as the basic EC and EP systems costs.

Similar overall weight and cost effects are shown for a typical advanced strike fighter in Figure 10. In this figure the effects of engine bleed and power extraction are also broken out. The "M.9 cruise" line assumes a level of engine degradation typical of subsonic cruise (shown in Figure 3) to occur throughout the entire flight, while the "M1.6 cruise" line similarly assumes typical supersonic values. These results are conservative since the peak power extraction will not usually be needed for a large percentage of the total flight time, but they do serve to bound the magnitude of the total engine-related effects.

The data of Figures 8 and 9 are for specific aircraft characteristics. However, overall penalties for power extraction are significantly influenced by varying aircraft performance levels. To illustrate these effects, the aircraft shown in Figure 11 were considered. They range from very high maneuverability levels (T/W = 120, W/S = 40) to relatively modest ones. The F-16, for example, is in the approximate middle of the range.

Since the effects of power requirements on aircraft size and cost tend to be reasonably linear over the ranges of interest, it is possible to plot the increment in fleet life-cycle cost per added kW of power extracted, as shown in Figure 12. For a fleet of 700 aircraft, the overall life-cycle cost increment ranges from approximately $70 million per kW for the highest-performance configuration to roughly $38 million per kW for the lowest. Ten kW of avionics power extraction will typically cost around $0.5 billion.

Having established the importance of power and cooling technologies in the overall scheme of things, the next question that arises is "where will technology developments be most effective in improving overall aircraft characteristics?"

To address this question, a series of aircraft with increasing performance, as defined in Figure 13, were sized. They range from a relatively low-performance fighter in the 15,000-lb class to a very-high-performance configuration of slightly less than 40,000 lb.

The incremental fleet life-cycle cost per kW of power extraction for these aircraft is depicted in Figure 14 under various assumptions about the level of power and cooling systems technology. The baseline levels assumed previously are shown at the top. The next lower line assumes that the unit <u>cost</u> of each system is reduced by one half, with no change in the unit weight of the system.

If the <u>weight</u> is halved but the cost held fixed, the next lower line results. It is clear that weight has a much greater <u>overall</u> cost impact than cost itself. This effect is considerably more pronounced at the higher performance levels - the direction in which aircraft design tends to move. Both weight and cost reduction have an additive effect that is essentially linear.

If both weight and cost improvements could in fact be attained, it would be possible to save approximately $29 million per kW in fleet life-cycle costs on the highest-performance aircraft, or about $290 million for a 10-kW power dissipation level. This is an order-of-magnitude measure of the level of research funding which could be expended on a "break-even" basis to attain the desired level of benefit <u>on this one aircraft type</u>. (This estimate is probably somewhat high because, in the real world, compromises would probably be made between cost and performance, so that the aircraft with the best EC/EP systems would tend to have higher performance levels than the baseline and somewhat lower cost differences than those stated. Of course, the higher performance levels would also have operational payoffs.) In any event, it is demonstrably worthwhile to pursue vigorous technology-development programs in these areas.

4. CONCLUSIONS

The development of improved EP/EC subsystems has been shown to be a potentially important contributor to continued improvement in overall aircraft cost and performance. Other studies (Reference 1, for example) have demonstrated the importance of good equipment thermal design in attaining these goals. It is also clearly important to establish good estimates of actual system requirements early in the design cycle to avoid carrying excess capacity. Although growth capability is obviously desirable, it is clearly not without cost.

In system development efforts, at least for the class of aircraft considered here, weight gains will tend to have a higher relative payoff than cost improvements, although both should obviously be pursued.

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APPENDIX

The aircraft gross weight is assumed to be made up of the following components

$$W_g = W_r + W_s + W_w + W_e + W_v + W_{Mis} + W_f + W_{EL} + W_{ECS} \quad (1)$$

where

W_r = Fixed weight items (fuselage, crew and crew station, etc.)

W_s = Weight of expendable stores

W_w = Weight that scales with wing area (wing, horizontal tail, certain controls)

W_e = Weight that scales with engine thrust

W_v = Weight that scales with required internal volume (fuel, payload, systems)

W_{Mis} = Miscellaneous weight items that scale with gross weight

W_f = Fuel weight

W_{EL} = Electrical power system weight which varies with power requirement

W_{ECS} = Environmental control (cooling, etc.) system weight.

It is further assumed that

$$W_w = \hat{S} S \quad (2)$$

where $\hat{S}$ is a fixed unit weight and S is the aircraft wing area and is determined by

$$S = \frac{W_g}{(W/S)} \quad (3)$$

where W/S is the desired wing loading. Similarly, for the engine thrust, T

$$W_e = \hat{E}\, T \quad (4)$$

$$T = \frac{T}{w}\,(W_g) \quad (5)$$

The volume-dependent weight is assumed to be proportional to the required volume. In this case

$$W_v = K_v(V_f + V_{EL} + V_{ECS}) \quad (6)$$

Where V_f, V_{EL}, and V_{ECS} are fuel, electrical power, and environmental control system volumes respectively. Equivalently, in terms of the weight and average densities (ρ) of these items

$$W = K_t\, W_f + \frac{\rho_f}{\rho_{EL}}\, W_{EL} + \frac{\rho_f}{\rho_{ECS}}\, W_{ECS} \quad (7)$$

The aircraft fuel fraction (i.e., the percentage of fuel weight to gross weight) is determined by range and combat requirements and is assumed to be fixed, i.e.,

$$W_f = \hat{W}_f\, W_g \quad (8)$$

where W_f is a fixed unit weight. The miscellaneous weight items are also assumed to be proportional to aircraft gross weight.

$$W_{Mis} = K_{Mis}\, W_g \quad (9)$$

The gross weight from equation (1) then becomes

$$W_g = \frac{W_r + W_s + \left(1 + K_t \frac{\rho_f}{\rho_{ECS}}\right) W_{ECS} + \left(1 + K_t \frac{\rho_f}{\rho_{EL}}\right) W_{EL}}{\left[1 - \frac{\hat{S}}{(W/S)} - \hat{E}\left(\frac{T}{W}\right) - (1 + K_t)\, \hat{W}_f - K_{Mis}\right]} \quad (10)$$

Electrical power and environmental control system weights are assumed to be proportional to their respective power requirements

$$W_{ECS} = \hat{W}_{ECS} (KW_o + \quad KVA) \tag{11}$$

Where

KW_o = Cooling load from nonavionics sources

KVA = Electrical power system capacity

η = Fraction of avionics system power capacity that must be dissipated by the cooling system

Similarly, the electrical power system weight is assumed to be proportional to the required power level

$$W_{EL} = \hat{W}_{EL} (KVA) \tag{12}$$

First-unit aircraft costs are broken down as

$$\$_{AC1} = \$_r + \$_e + \$_{AV} + \$_{EL} + \$_{ECS} \tag{13}$$

where

$\$_r$ = Cost of everything but engine, stores, electrical, environmental control, and avionics systems

$\$_e$ = Cost of engines

$\$_{AV}$ = Cost of avionics

$\$_{EL}$ = Cost of electrical power system

$\$_{ECS}$ = Cost of environmental control system

Cost estimating relationships are assumed of the form

$$\$_r = \hat{\$}_r (W_g - W_s - W_e - W_f)^{\hat{r}} \tag{14}$$

$$\$_e = \hat{\$}_e T^{\hat{e}} \tag{15}$$

$$\$_{AV} = \text{Fixed} \tag{16}$$

$$\$_{EL} = \hat{\$}_{EL} W_{EL} \tag{17}$$

$$\$_{ECS} = \hat{\$}_{ECS} W_{ECS}^{\widehat{ECS}} \tag{18}$$

RDT&E costs are represented as

$$\$_{RDTE} = \$_{RTA} + \$_{RTAV} + \$_{RTP} \tag{19}$$

where $\$_{RTAV}$ is a fixed "avionics" RDT&E cost and is assumed to include both EP and EC *systems. Airframe and engine RDT&E costs* are given by, respectively

$$\$_{RTA} = \hat{\$}_{RTA} (W_g - W_s - W_f)^{\widehat{RTA}} \tag{20}$$

$$\$_{RTP} = \hat{\$}_{RTP} T^{\widehat{RTP}} \tag{21}$$

Operations costs are approximated as

$$\$_{OPS} = \left(K_{OAC} \$_{ACT} + \$_{OPC} N + \$_f N\right)\left(\frac{Yr}{10}\right) \tag{22}$$

where $\$_{OPC}$ is a "fixed" operations cost per unit aircraft (determined by curve fits of data from several sources), N is the number of aircraft to be produced, and Yr is the average number of years of operation for each aircraft. K_{OAC} is a proportionately constant between operations costs and aircraft procurement costs and accounts for size effects, maintenance effects, etc. Also, the procurement costs over N units is

$$\$_{ACT} = \frac{(\$_r + \$_e + \$_{AV})N^{1+b}}{1 + b} + \frac{(\$_{EL} + \$_{ECS})^{.848}}{.848} \tag{23}$$

where

$$b = \frac{\ln F}{\ln 2}$$

F = a "learning-curve" factor (reduction in cost each time the experience doubles)

The total aircraft fleet life-cycle cost is then

$$\$_T = \$_{ACT} + \$_{OPS} + \$_{RDTE} \tag{24}$$

Typical constants for advanced fighters are given in Table 1.

Table 1

TYPICAL AIRCRAFT SIZING CONSTANTS

CONSTANT	VALUE	CONSTANT	VALUE
W_s	Variable	$\hat{RTA}$	1
W_r	4700	$\hat{\$}_f$	80
K_t	0.14	$\$_{OPC}$	4.88E6
$\hat{S}$	8.6	Yr	10
$\hat{E}$	0.18	K_{OAC}	0.383
K_{Mis}	0.1	$\hat{W}_{ECS}$	20
$\$_{AV}$	3.8×10^6	η	0.2
$\hat{\$}_r$	3.44×10^4	$\hat{W}_{EL}$	4
$\hat{R}$	0.603	$\hat{\$}_{ECS}$	1720
$\hat{\$}_e$	2900	KW_o	5
$\hat{e}$	0.694	ρ_f/ρ_{ECS}	2.90
$\hat{\$}_{EL}$	390	ρ_f/ρ_{EL}	1
F	0.87	$\hat{ECS}$	0.817
N	Variable	$\hat{W}_f$	.3(variable)
$\hat{\$}_{RTP}$	5.93E6	T/W	Variable
$\hat{RTP}$	0.435	W/S	Variable
$\$_{RTAV}$	100E6	$\hat{\$}_{RTA}$	60700

REFERENCE

1. Anon. 1974, "F-16 System Design Trade Study Report, Vol. 29, Environmental Control Subsystem Design", General Dynamics Report FZM-401-147, Fort Worth, Texas.

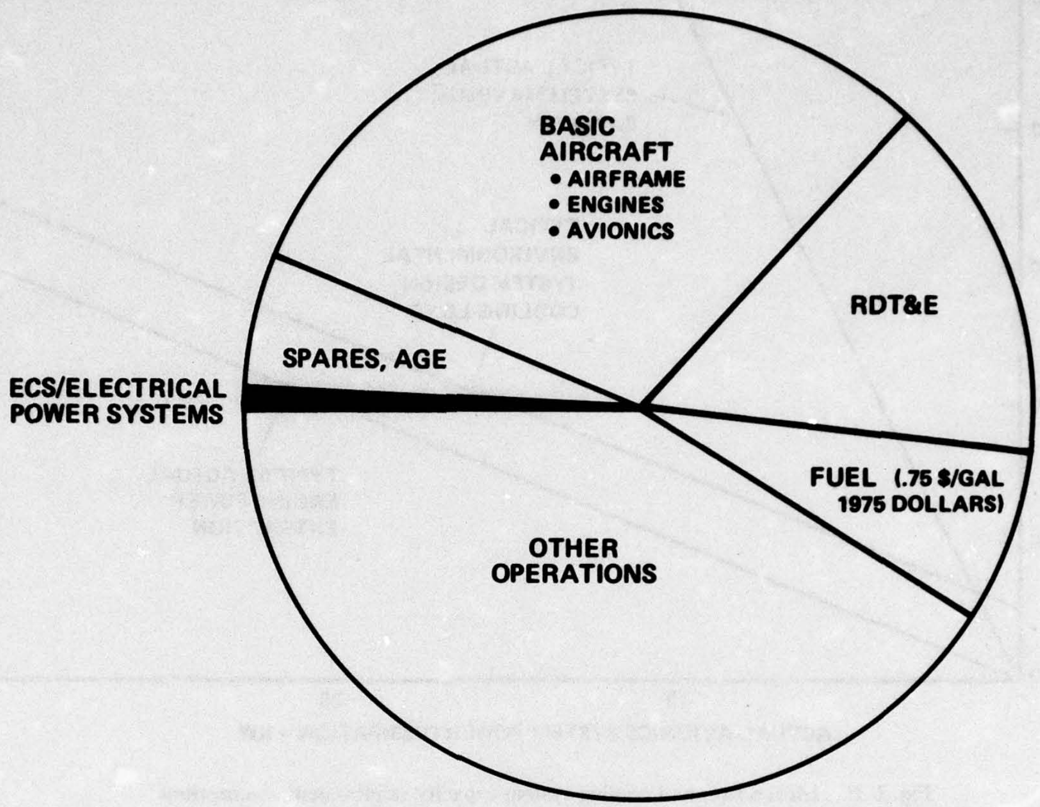

Fig. 1 Typical distribution of an aircraft's costs over it's life cycle

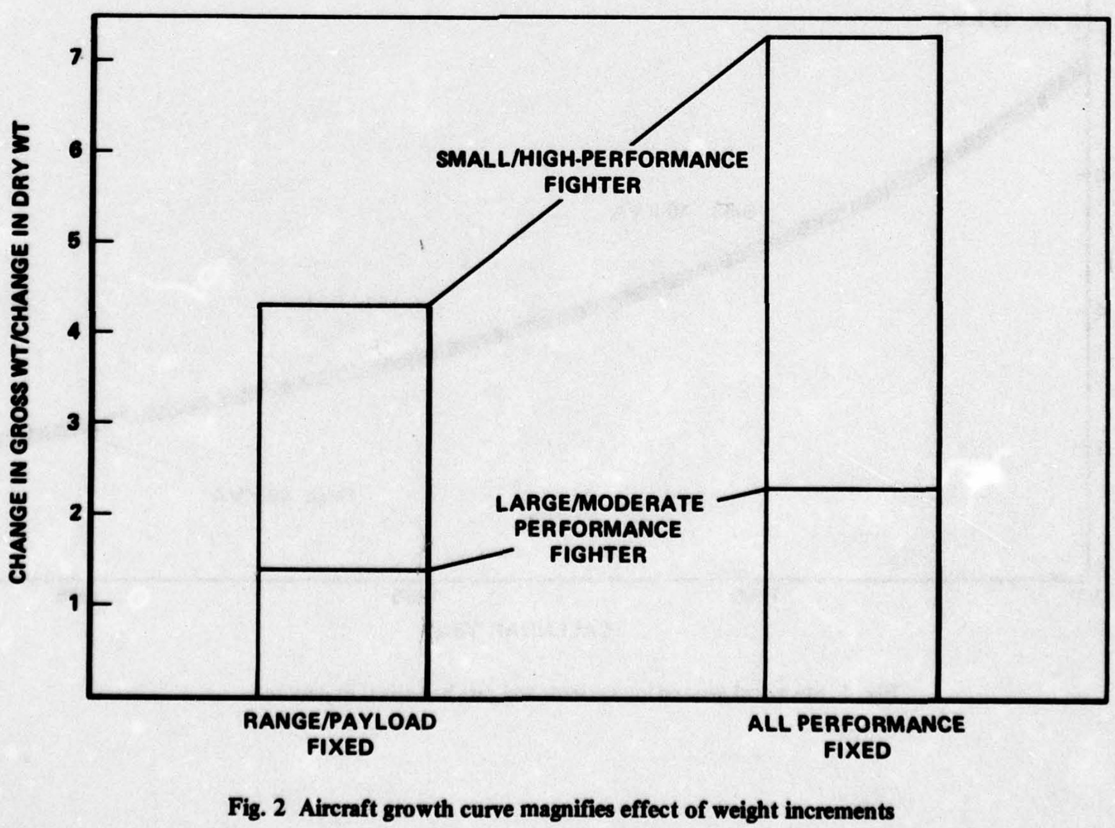

Fig. 2 Aircraft growth curve magnifies effect of weight increments

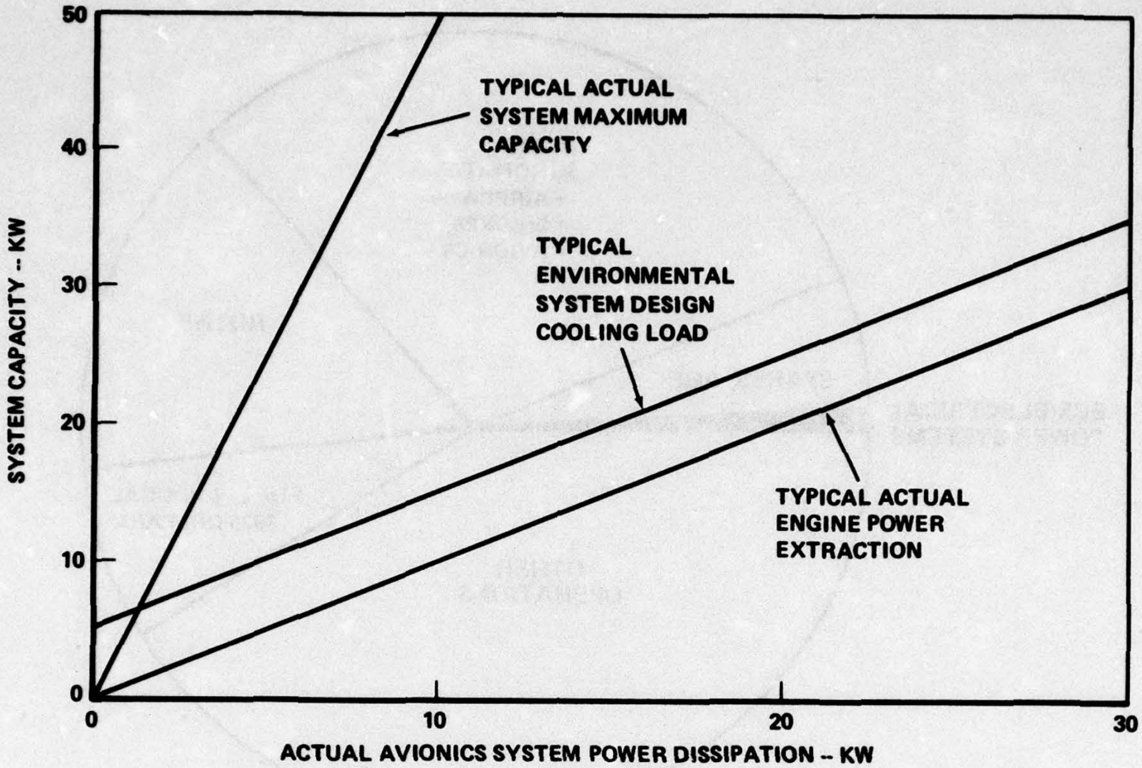

Fig. 3 Electrical power and cooling system capacity requirement assumptions

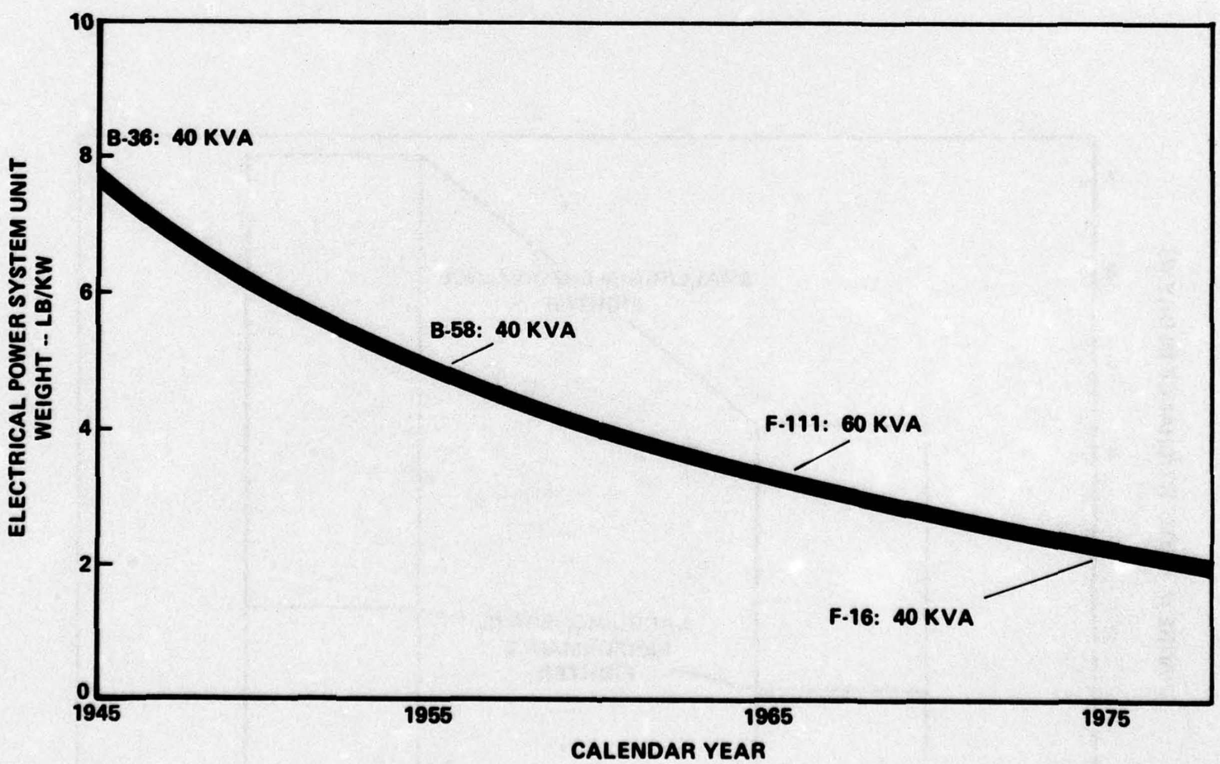

Fig. 4 Electrical generating systems weight: historical experience

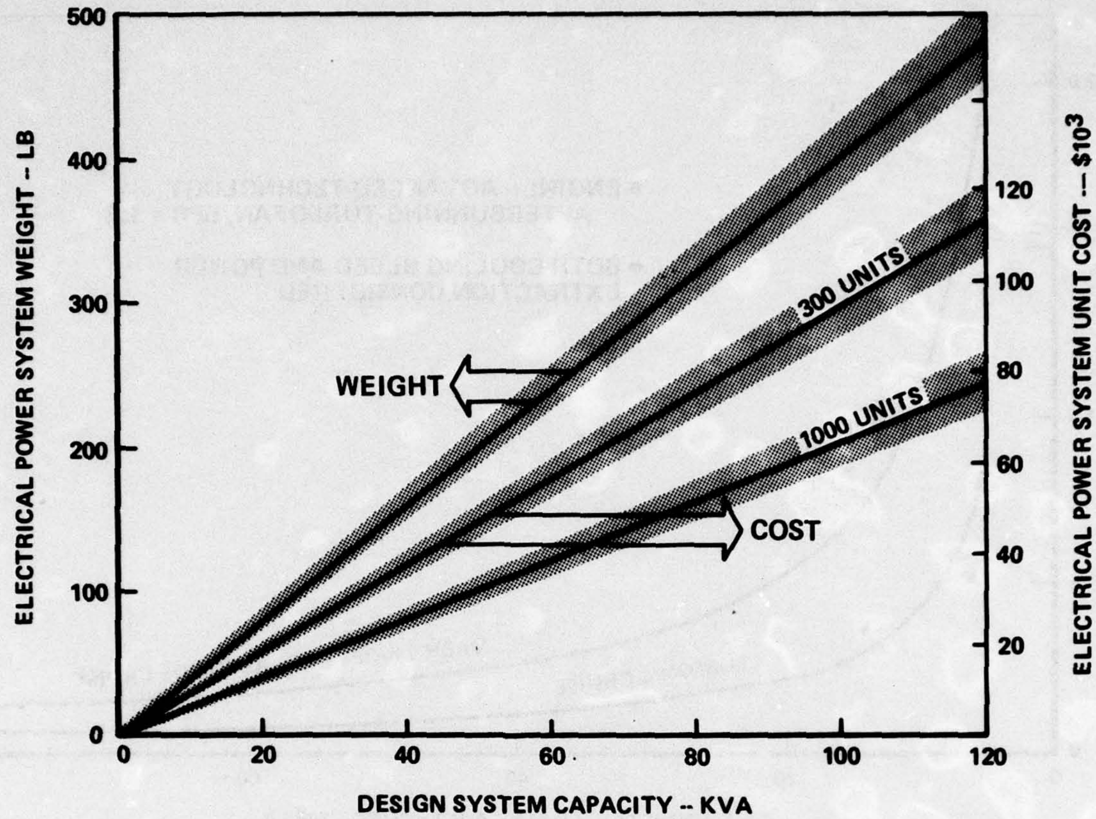

Fig. 5 Weight and cost of electrical power supply systems

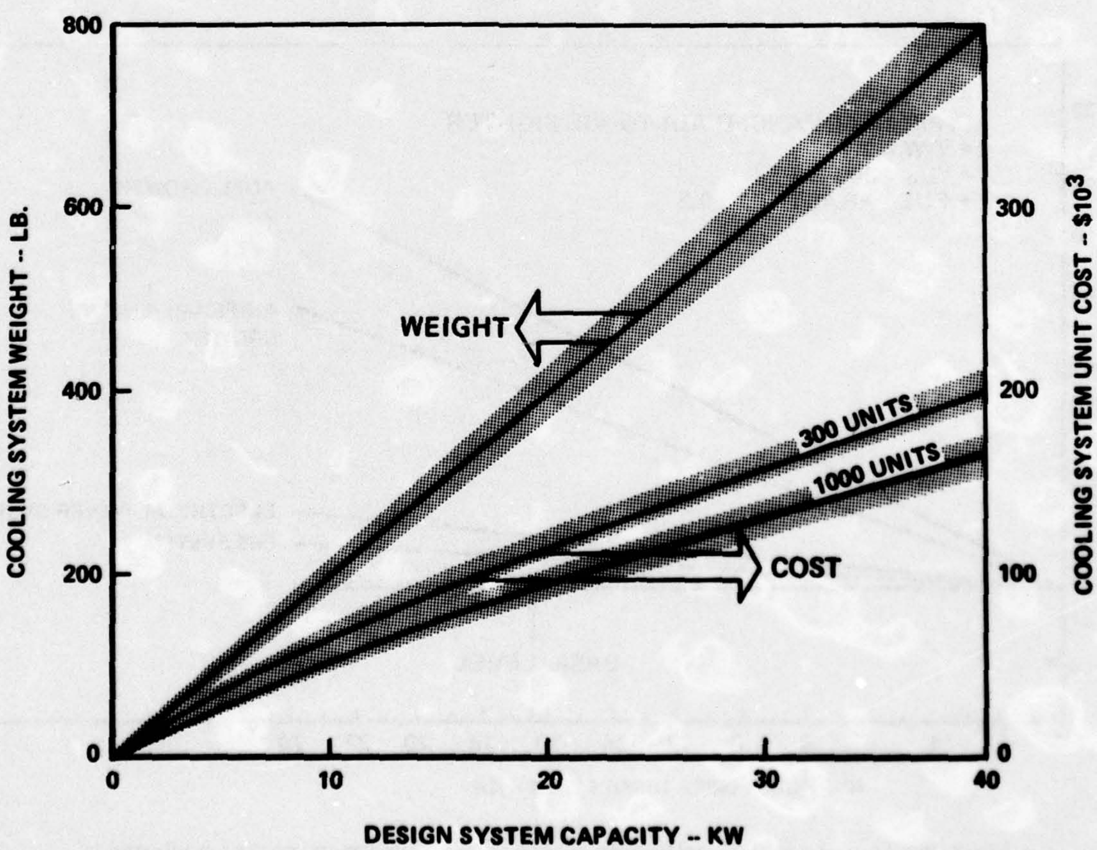

Fig. 6 Weight and cost of cooling systems

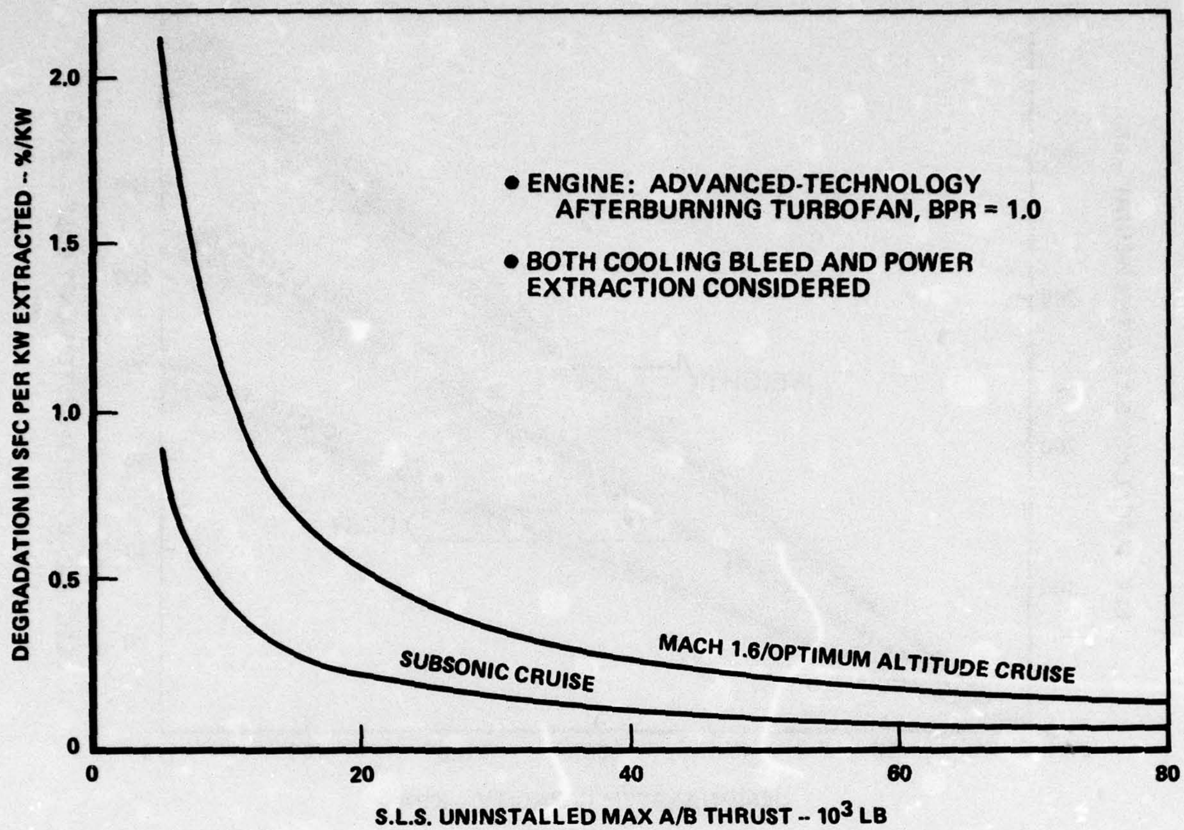

Fig. 7 Typical degradation in engine specific fuel consumption per kilowatt extracted

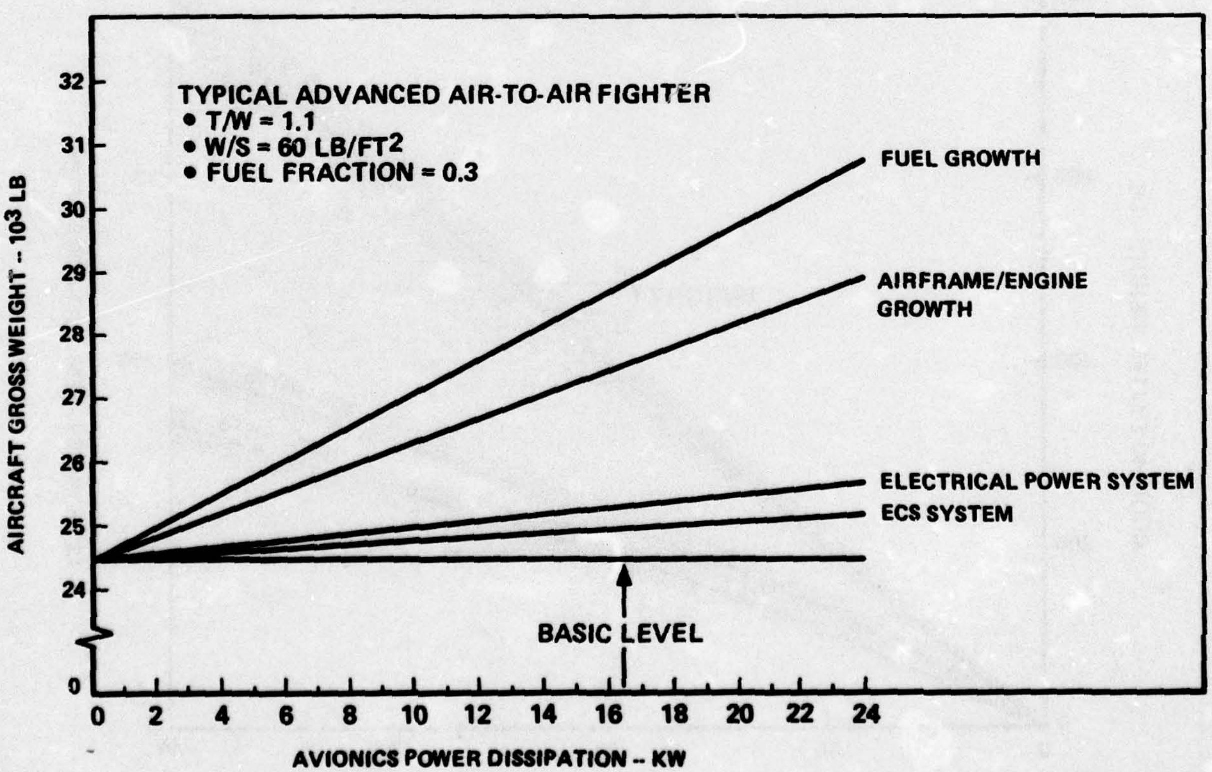

Fig. 8 Breakout of overall aircraft weight growth caused by power and cooling requirements

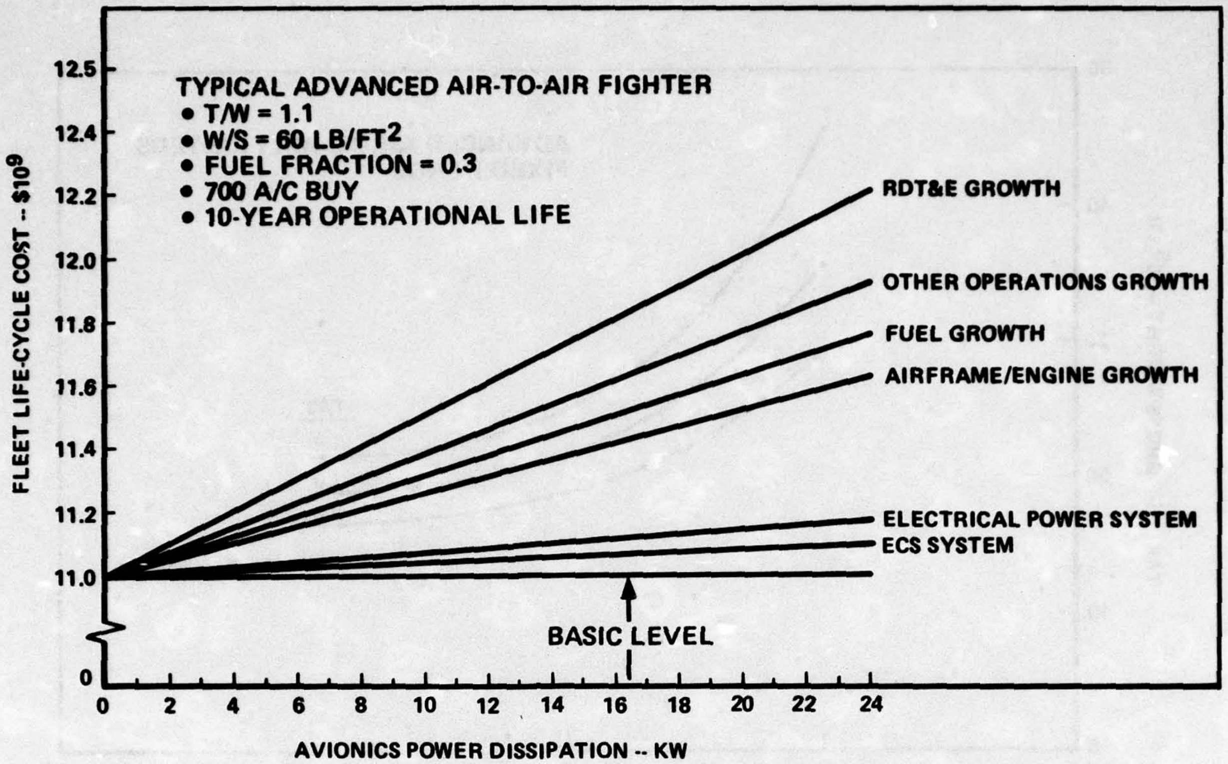

Fig. 9 Breakout of overall cost growth caused by power and cooling requirements

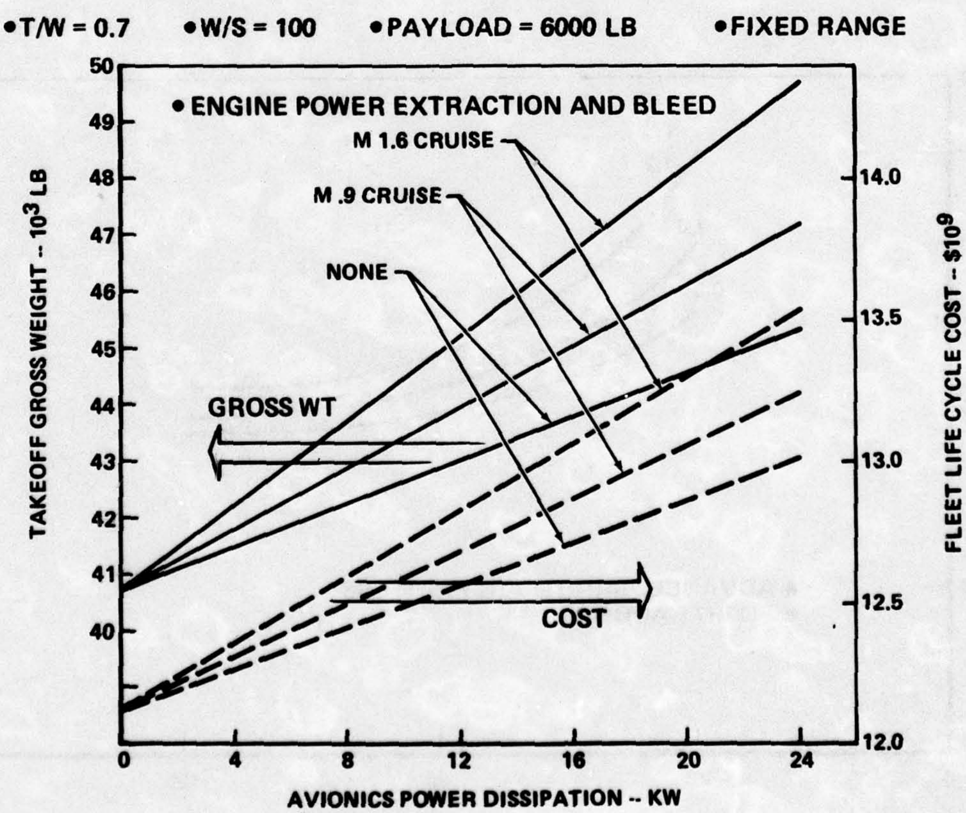

Fig. 10 Effect of engine bleed and power extraction on aircraft weight and cost

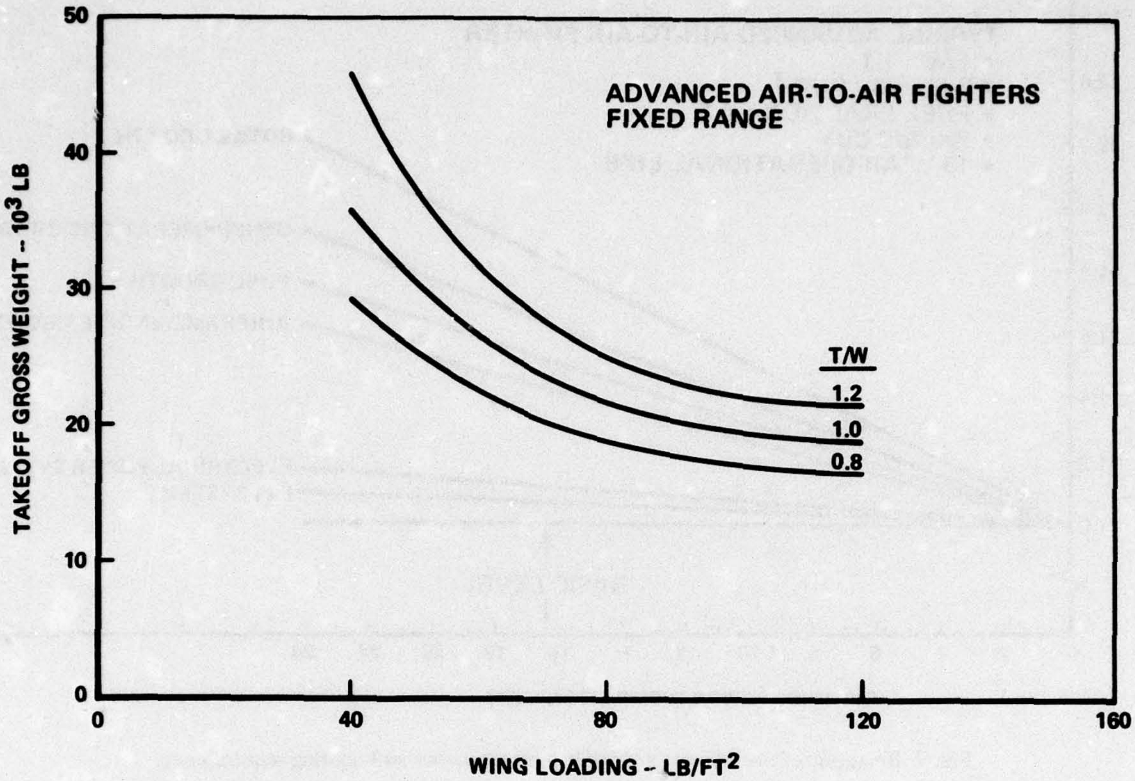

Fig. 11 Effect of fighter performance on takeoff gross weight

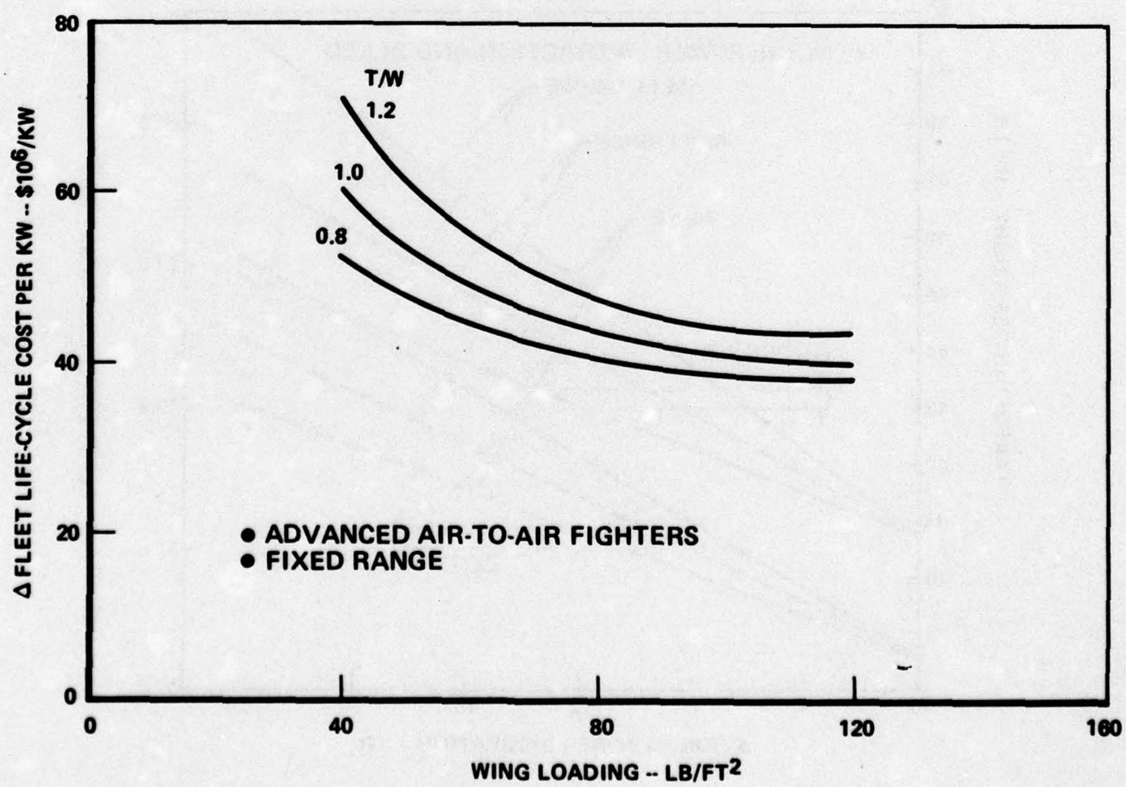

Fig. 12 Effect of power and cooling requirements on fleet life-cycle cost

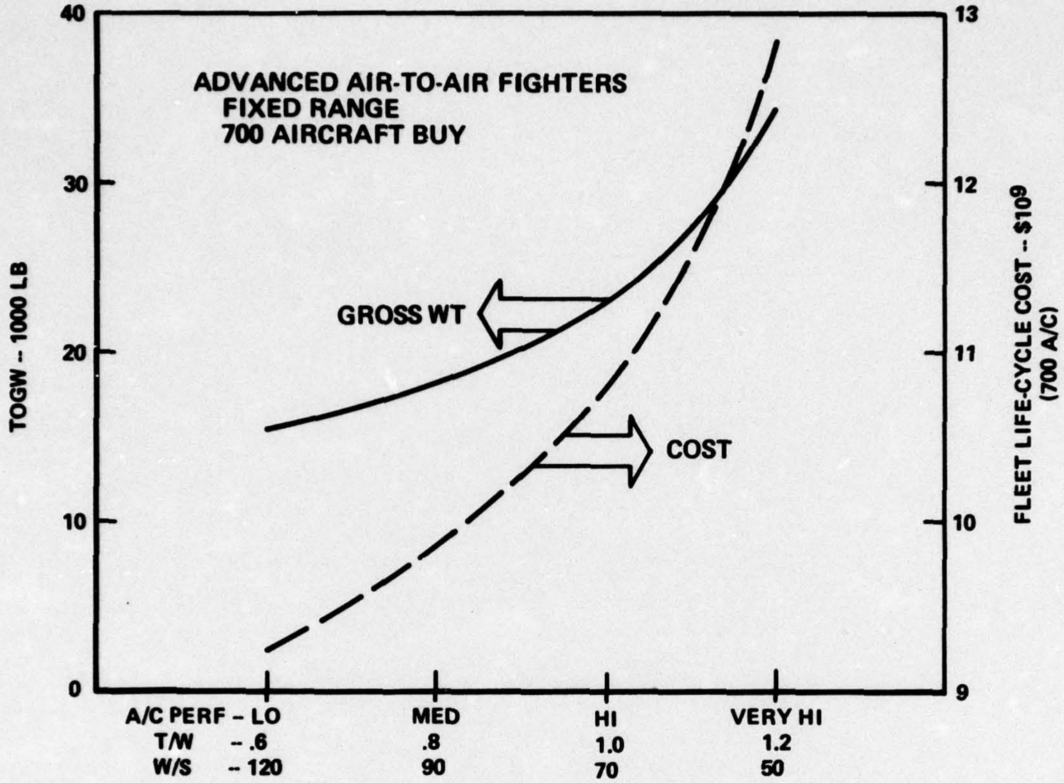

Fig. 13 Aircraft characteristics assumed for technology trades

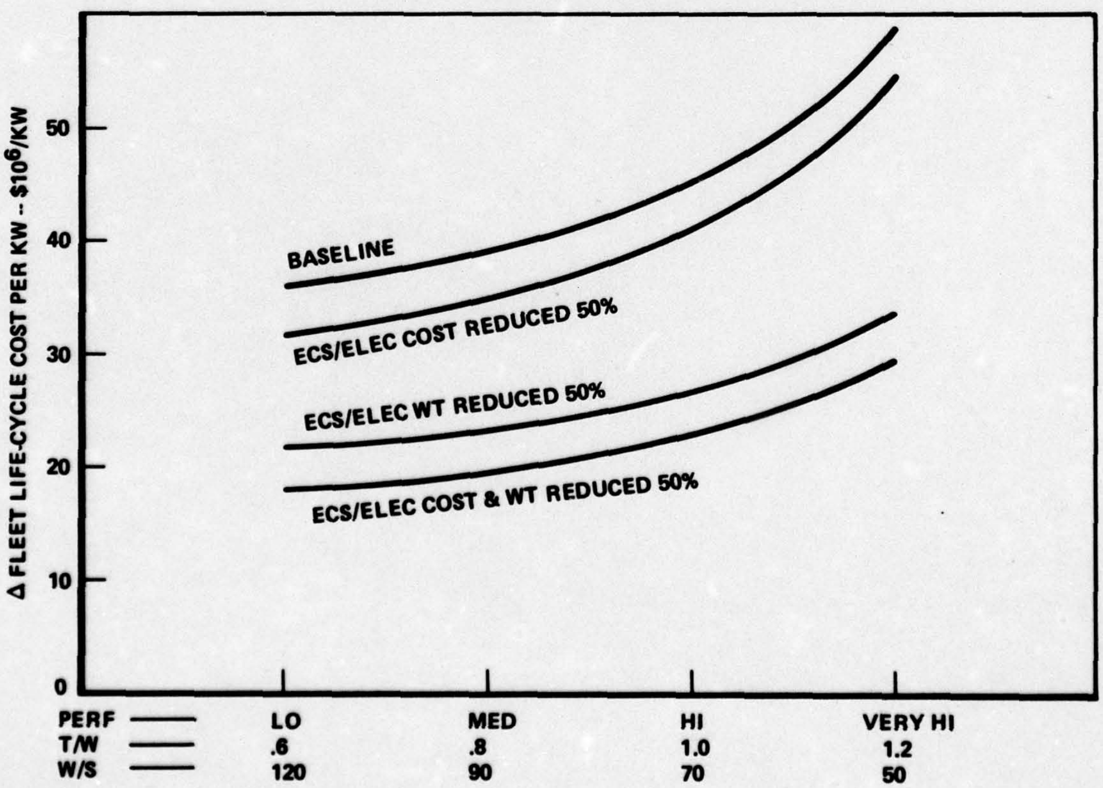

Fig. 14 Effect of technology improvements on life-cycle cost

COOLING OF ELECTRONIC EQUIPMENT IN RELATION TO COMPONENT TEMPERATURE LIMITATIONS AND RELIABILITY

By

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SUMMARY

The rapid growth in the application of electronics in military aircraft has created a situation in which the heat dissipated by electronic equipments can impose a penalising demand on the aircraft for cooling supplies if the requirements for equipment reliability are to be met. Basically the problem is one of maintaining the component parts of an equipment at a level of temperature consistent with performance and reliability. This paper reviews some of the problems and the cooling techniques affecting avionic equipment design.

The temperature limitations of electronic components are discussed together with the influence of temperature on the reliability of the equipment and its component parts. The thermal performance of two representative avionic designs are compared and the influence of continuing solid state device developments on future equipment heat loads is considered. The limitations of air as a heat transfer medium are examined and a comparison made with liquid cooling.

Concluding comments highlight some of the factors which will undoubtedly influence developments in the cooling of future avionic systems. The importance of sound thermal design is stressed and a recommendation is made that future equipment design specifications should include a better definition of the actual temperature environment for the purpose of design and reliability.

INTRODUCTION

The temperature conditions in an aircraft are one of the more significant operational environments affecting avionic equipment reliability. Operation under conditions that result in an excessively high temperature for the component parts can seriously degrade this reliability. In an aircraft these high temperatures can arise from several causes such as a high bay temperature, deployment in a hot climate, a large avionic heat load, inadequate cooling supplies, unsatisfactory thermal design, etc.

The cooling of electronic equipment is basically a twofold problem. To the equipment designer it is one of transferring heat from power dissipating components and parts to a suitable sink, either the aircraft atmosphere or its cooling services, so that the temperature of all components whether heat dissipating or not, are maintained within a range compatible with performance and reliability. To the aircraft designer it is one of providing an aircraft atmosphere or cooling service compatible with the equipment's needs.

The means of achieving acceptable thermal performance depends largely on the type of equipment and the function it has to perform. An equipment containing a large number of components each dissipating a small amount of power (eg a computer containing many integrated circuits mounted on boards) presents on entirely different type of design problem than does a high powered radar or jammer where heat dissipation is high and from several concentrated sources namely the transmitting components. Many other factors can influence thermal performance, for example, the constructional features of the equipment whether pressurised enclosed or of unsealed design and also the location of the equipment within an aircraft whether in a pressurised or unpressurised zone.

This paper reviews some of the problems and techniques of cooling electronic equipment, the emphasis being given to systems using air as the cooling medium.

COMPONENT DEVELOPMENT AND POWER DISSIPATION

The development of solid state technology over the last two decades has resulted in a major reduction in the electrical power required to carry out a given circuit function and it has accelerated the process of miniaturisation. These trends and the application of digital data processing has greatly extended the use of electronics in aircraft.

Avionic equipments can be broadly classified into two main functions, data acquisition and data processing. It is in the field of data processing that solid state technology has had most impact. Acquisition functions, particularly high powered RF equipments, have seen no corresponding improvements and the thermal efficiency of RF power generating components has shown little change. These components are the major sources of waste heat in many military aircraft. The overall effect is that avionic heat loads over the past two decades have shown an increasing growth pattern, and although new solid state device developments such as CMOS technology, low power Schottky, etc may result in lower power dissipation, the potential for even greater miniaturisation gives no reason to suppose that the future growth pattern will show a marked change.

ELECTRONIC COMPONENT LIMITATIONS

The temperature boundaries which are significant in the operation of an electronic equipment are shown diagrammatically in figure 1. The equipment must be capable of functioning over the temperature range of the operational environment, as indicated by the hatched area to the left of the diagram. The component parts should never at any time exceed their safe upper limit temperature, represented by the hatched area on the right. Improved reliability is obtained by de-rating as shown in the centre of the diagram. The temperature range between the environmental boundary and the derated limits is that in which the temperature gradients needed to carry out the cooling process must be fitted.

A list of components typically used in avionic equipments together with their normally accepted upper temperature limits is given in Table 1. These limit values are either the internal hot spot temperature, the case temperature, or the maximum ambient in which the component can safely operate; the ambient or case temperature are the ones normally specified by the component manufacturer. For semi-conductor devices the hot spot temperature relates to the junction whereas the case or ambient temperature is determined by the electrical loading.

With few exceptions it can be seen that the maximum ambient air or case temperature at which components can safely operate is within a range of about +85 to +125°C. These temperatures are maximum limits and prolonged operation at these levels would seriously affect reliability.

EFFECTS OF TEMPERATURE ON RELIABILITY

The data in table 2 gives examples of typical failure rates related to temperature for some commonly used electronic components; the temperature reference is that of the component surface, the data source is MIL-STAN-217B. Figure 2 is the same data processed and presented in graphical form to show relative failure rate against temperature, using +25°C as the reference. The slope of each curve shows that the failure rate progressively increases with increase in temperature, each component having a different characteristic. A bipolar digital device for example shows little change with increasing temperature whereas a bipolar linear device or hybrid integrated circuit exhibits a large change.

The significance of these failure rate characteristics on the overall reliability of an equipment will depend upon the types of components used and the quantities involved and a component which shows a large characteristic change may have little overall effect if it is used only in small quantities. A curve produced by the reliability group in one of the UK Defence Establishments is shown in figure 3, this indicates the change in the relative failure rate of military electronic equipments with temperature. It has been derived from published failure rate data for the component parts and is representative of a component population typical of many avionic equipments; experimental and service experience has largely substantiated this characteristic pattern. The shape of the curve below -30°C and above +80°C shown dotted, is less certain since it has been extrapolated from available information.

The temperature relates to the mean internal ambient, that is the temperature of the air in which the components are operating. It can be seen that optimum reliability is achieved when this temperature lies between 10 and 20°C but for all practical purposes a range -20°C to +50°C would be acceptable.

The temperature conditions in most military aircraft will vary over a wide range determined by the external environment, the temperature and quantity of the cooling supplies, the power dissipation within the aircraft and the aircraft's operational state at the time. The effect of these temperature changes on reliability depends upon the accumulated time at different levels of internal temperature and provided an equipment is maintained for most of its operating life with the internal ambient within the range of optimum reliability (ie -20 to +50°C) occasional excursions of temperature up to the limit levels for the components may have little overall effect.

For many equipments the cooling air provided by the aircraft is the most critical factor determining internal temperatures. The flow rate and temperature of this cooling air will vary widely in service use depending upon engine speed, flight speed, outside air temperature, altitude, etc. and in a high performance aircraft it is not uncommon for the mass flow to change by a factor up to times 3 or more and the temperature by +20 or +30°C. The more adverse cooling situations often occur for limited periods for example, when the aircraft engines are idling (during taxying, awaiting take-off, or in an idle descent) or when the aircraft is being serviced on the ground in a hot climate. In order to avoid excessive equipment temperatures it is important that these phases of operation should be restricted to short periods and they should not accumulate to a significant proportion of the equipment's operating life.

EQUIPMENT DESIGN IN RELATION TO COMPONENT TEMPERATURE

The actual temperature attained by a component within an equipment is largely determined by the effectiveness of internal heat transfer processes. Figure 4 shows in diagrammatic form two types of construction often used for avionic equipments. The diagram marked (a) represents a convection cooled unit where the components are mounted on boards and the heat dissipated is partly conducted but largely convected to the equipment cover which in turn is cooled by air passing over the outside surface. The interfaces are numerous and thermal impedance particularly at surface to air interfaces can be large resulting in a high component temperature, as shown in the temperature gradient diagram. The equipment represented by (b) is a design achieving more effective cooling, the heat from the component(s) is transferred by a designed conduction path directly to a cold plate integral with the equipment. Thermal impedance is greatly reduced and a number of surface to air interfaces, normally producing large gradients, are avoided. This construction is representative of an equipment where particular attention has been given to thermal design using for example metal cored pcb's or boards fitted with thick copper strips which in turn are thermally coupled to the wall of a cold plate. Internal forced air circulation in place of the designed conduction paths can often achieve similar standards of thermal performance.

A point worth stressing which is a common misconception is a belief that an increase in the quantity of cooling air will give a corresponding decrease in the component temperatures. The effect is to reduce only the temperature gradients external to the equipment, gradients within the equipment are unchanged and the overall effect is often a marginal reduction in the temperature of the components.

The cooling of an equipment is also dependant on the external area scrubbed by the cooling air. In many aircraft installations equipments are closely packaged and air movement over a large proportion of the external surface is impeded. In long narrow aircraft bays the air passing through the bay can tend to laminar flow with inadequate turbulence. Both these effects can reduce cooling and raise component temperatures.

FACTORS AFFECTING COOLING REQUIREMENTS

The cooling requirements for an avionic equipment are determined by a number of factors including

a the operating temperatures of component parts for safe and reliable performance.

b the amount of heat dissipated.

c the temperature and type of coolant.

The operating temperature limits of the component parts are determined by the materials used and the author is unaware of any technological developments that will significantly change these present limits. What may be of significance however is the development of low power dissipating devices such as CMOS and the influence of L.S.I technology. Whether these techniques will have any impact in arresting or reversing the trends in avionic heat loads, is one of conjecture. Data from a recent survey in the UK covering a limited range of avionic units would indicate a potential for increasing growth. Figure 5 plots the overall power density for the units covered by the survey against the design date; eleven of the units use TTL/DTL technology, the other two a proportion of CMOS. Although the sample size is small the trend is marked and shows increasing power density for the same technology. The two units employing CMOS devices have approximately maintained the power density of the 1967 vintage TTL/DTL units.

No firm conclusions can be drawn from these survey results which may simply indicate a continuing trend to increased packaging density and therefore greater miniaturisation. However, should the space thus saved be utilised to accommodate more electronics, the implication is a continuing growth in the heat load.

EQUIPMENT COOLING

The problem of cooling equipment is not only concerned with the transfer of heat within the equipment but also of transferring this heat from the equipment and rejecting it from the aircraft. To-date except for a few liquid cooled systems, air has been the medium used for cooling avionic equipment. It has several disadvantages: a relatively low specific heat, low density at sea level and even lower density at altitude. The removal of 1kW of heat with a mass air flow of 1kg/min will raise the temperature of the cooling air by about $+60^{\circ}C$ and this is coupled to a volumetric flow at sea level of about 0.8 M^3/minute.

Figure 6 is included to show the cooling performance achieved in some current radar/avionic equipments where except for one, liquid cooled unit air is the coolant. The temperature rise, the horizontal scale, is that of the surface of selected critical components above the coolant inlet temperature; the quantity of cooling air is given on the vertical scale and is expressed as the flow rate per unit power dissipation. The equipments included are generally considered to be of a good thermal design standard, using either internal forced air circulation or conduction path cooling in conjunction with a heat exchanger or cold plate.

A number of observations can be made but it would be misleading to attempt to draw general conclusions since only a limited number of equipments have been considered and of these two are radar systems which include concentrated heat sources.

These observations are

1 there is an obvious exponential increase of component temperatures with reduced air flow as would be expected.

2 for the equipments considered, cooling air of flow rate lower than about 1.5 to 2 Kg/min/KW would result in high component temperatures which could seriously affect reliability. Even at this flow rate good thermal design is essential if component temperatures are not to be excessive.

3 Comparison of the two flight conditions covering the same radar equipment is of interest, increasing the air flow rate from 1.5 to 2.8 Kg/min/KW reduces component temperatures by about $10^{\circ}C$. The effect of wider changes in the flow rate have been estimated and are shown by the dotted line; an increase to 6Kg/min/KW is expected to reduce component temperatures a further $5^{\circ}C$ whereas reducing the flow rate from 1.5 to 0.75 Kg/min/KW would increase component temperatures by about $20^{\circ}C$. This supports the point made earlier that increasing the quantity of cooling air does not necessarily achieve a corresponding reduction in component temperatures. For this particular design of equipment there is little to be gained by increasing the cooling supplies above 3 Kg/min/KW.

COOLING BY LIQUID

Should avionic heat loads continue to rise and the packaging trends indicated by figure 5 result in equipment of higher power density, then the use of air as a coolant in high performance aircraft could become unacceptable or uneconomical. Liquids such as silicate ester based fluids and fluorochemicals have major advantages as coolants, and although the specific heat of these fluids is not significantly different from that of air, the density and conductivity characteristics make it possible to use smaller heat exchangers operating with a higher coefficient of performance. Figure 7 illustrates the relative effectiveness of a fluorochemical coolant as compared with natural and forced air convection; in its convective mode a fluorochemical liquid has a heat transfer coefficient at least one order better than air.

The use of liquid coolants in high performance aircraft has increased over the last 10 years with their application to high powered military radars where the heat flux density of the RF components has been so high as to make air cooling impracticable. In general liquids have been used only where it has been essential. It would however seem a logical step in those aircraft where a liquid system already exists, to extend its application to the major avionic systems. The sealing of quick release fluid couplings if used on a larger scale than hitherto may give rise to user problems although this no doubt could be resolved by development. However, the use of liquid cooled "cold plates" permanently installed in the aircraft could be an attractive alternative with the equipment clamped to the plate and cooled by conduction. This would eliminate the need for quick release couplings but would require the development of suitable clamping arrangements giving good thermal conduction.

COMMENTS AND CONCLUSIONS

1 It is difficult to predict the course of future developments in the cooling of aircraft equipment because of the many uncertainties. A number have been mentioned in this paper, for example the future growth of avionic heat loads, the impact of solid state developments and the need to maintain equipment temperatures at levels compatible with reliability. Additionally other influences which will have some bearing on these developments are the quality of aircraft power supplies, the impact of integrated packaging as outlined in the ARINC Installation Proposals, and the economics and penalties of providing large quantities of cooling air.

2 A high standard of thermal design could alleviate many cooling problems and should avionic heat loads continue to increase or even remain at present levels with air retained as the prime coolant, then this could be the only practicable solution for many avionic equipments.

In any event improvements in the standard of thermal design need to be actively encouraged if the demands on the aircraft cooling services are to be minimised. Ideally the requirements for thermal design should be given status in the Equipment Design Specification rather than the current practise by implication in the reliability and environmental requirements.

3 The time spent at different levels of temperature is an all important consideration for purpose of reliability. It is normal practise in the equipment design documents to specify only the design limit temperatures representing the more extreme conditions in which an equipment has to survive and operate. These conditions invariably have only a low probability of occurrence and are therefore of limited use for reliability development. In order that an equipment designer may have a better appreciation of the operating conditions the design specification should ideally include information indicating the actual temperature environment expressed on a probability basis.

4 Good thermal design can have implications on costs, weight and mechanical complexity and these need to be justified on the basis of cost and technical effectiveness. A danger is one of unnecessary sophistication in design for those equipments where effective cooling is possible by simple means. Many electronic equipments, particularly ground based installations make use of direct cooling where the air is blown over the components; cooling can be effective without elaboration of design. It is suggested that there are many avionic equipments where this simple form of cooling would be acceptable, particularly those installed in conditioned zones of an aircraft. The quality of the aircraft atmosphere and the cooling supplies and their effects on reliability may need investigating as would the validity of some of the requirements of general design specifications. In general these specifications imply an atmosphere unacceptable to exposed electronic parts for example in respect of salt, moisture, dust, fluid contamination etc. In many cases these requirements relate to tests which are either accelerated or time compressed and their effects can often be misleading.

	HOT SPOT TEMP.	MAX. CASE OR AMBIENT TEMP.
SEMI-CONDUCTOR DEVICES. Transistors (silicon)	150/200°C	typically 125°C
I.C's Hybrids	175°C	125°C
Monolithic	150°C	125°C
R.F. POWER DEVICES. Magnetrons, Klystrons, T.W.T.	120/150°C	
Gunn Diodes		70°C
Solid State Diodes	150/200°C	depends on load
DISPLAY Tubes (display & storage)		150°C
Solid State (L.E.D's)	100°C	
Scan Convertor		90°C
Vidicon	70°C	
MISC. Resistor Fixed Film	150°C	70°C (max. load)
Capacitors Ceramic		85, 125 or 150°C
Plastic and Glass		125°C
Tantulum		85 or 125°C
Electrolytic		85 or 125°C
Connectors	125, 150 or 200°C	$\geqslant$ 70°C max. load current

Table 1 Examples of Upper Temperature Limits for Electronic Components.

	Failures / 10^6 hrs				
	25	50	100	125	150°C
MICRO ELECTRONIC DEVICES					
1) Hybrids	1·8	4·4	18·2	33	53
2) Bipolar Digital	0·3	0·31	0·43	0·61	0·9
3) Bipolar Linear	0·33	0·37	1·45	4·3	16·8
TRANSISTORS F.E.T.	0·9	1·15	2·35	4·4	—
N.P.N.	0·48	0·63	1·27	2·18	—
RESISTOR Fixed Film	0·0085	0·01	0·0175	0·024	—
CAPACITORS Ceramic	0·018	0·05	0·37	—	—
Solid Tantulum	0·04	0·052	0·152	0·48	—
CONNECTORS	0·432	0·8	2·36	4·4	7·2

Table 2 Examples of Failure Rates of Electronic Components.

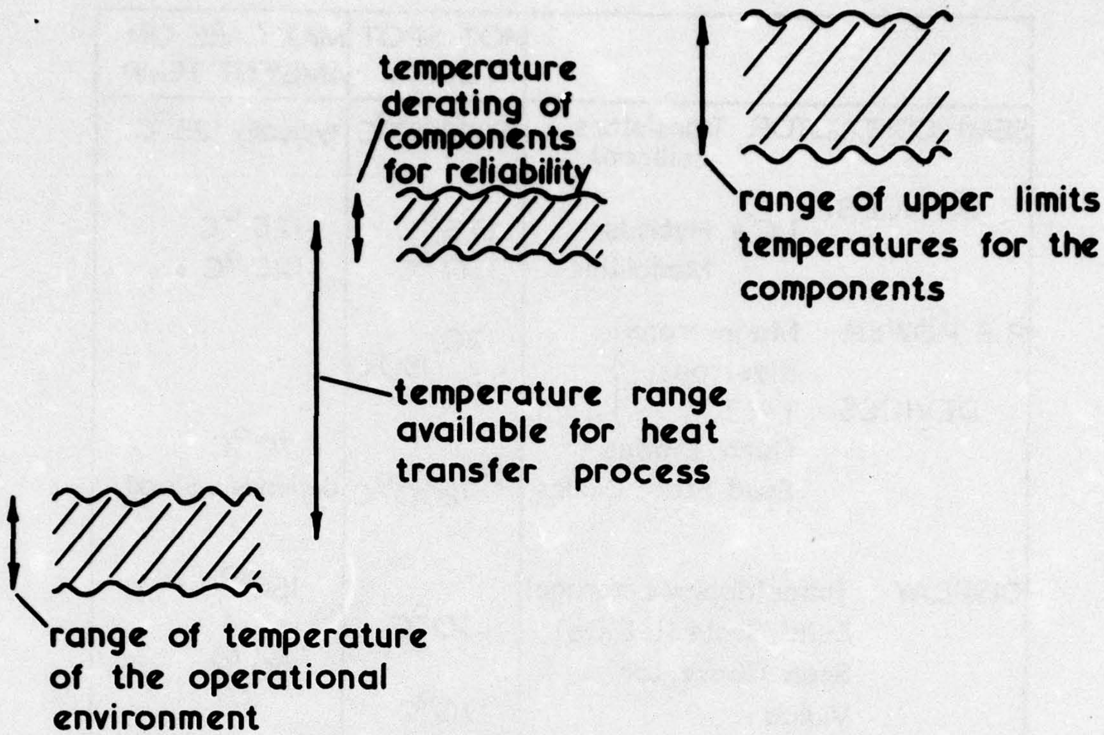

Figure 1 Operating Temperature Boundaries for Electronic Equipment.

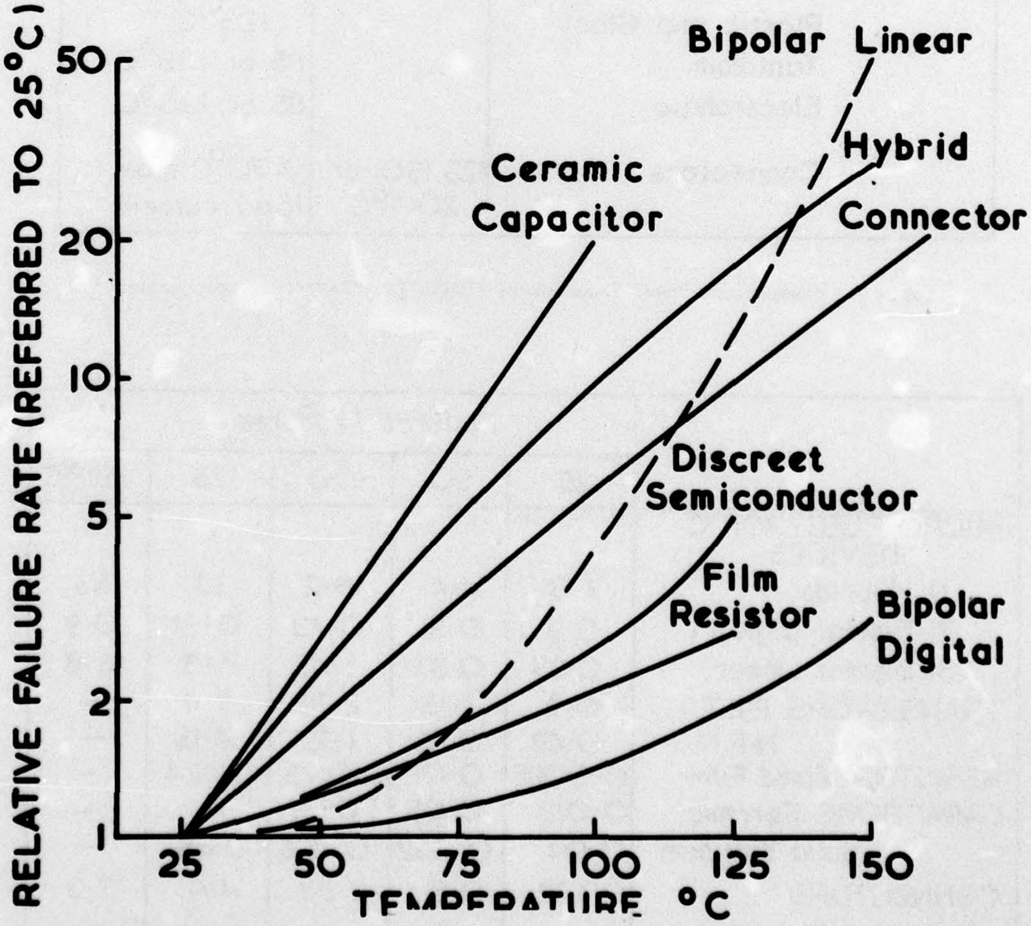

Figure 2 Effects of Temperature on Relation Failure Rate of Typical Avionic Components.

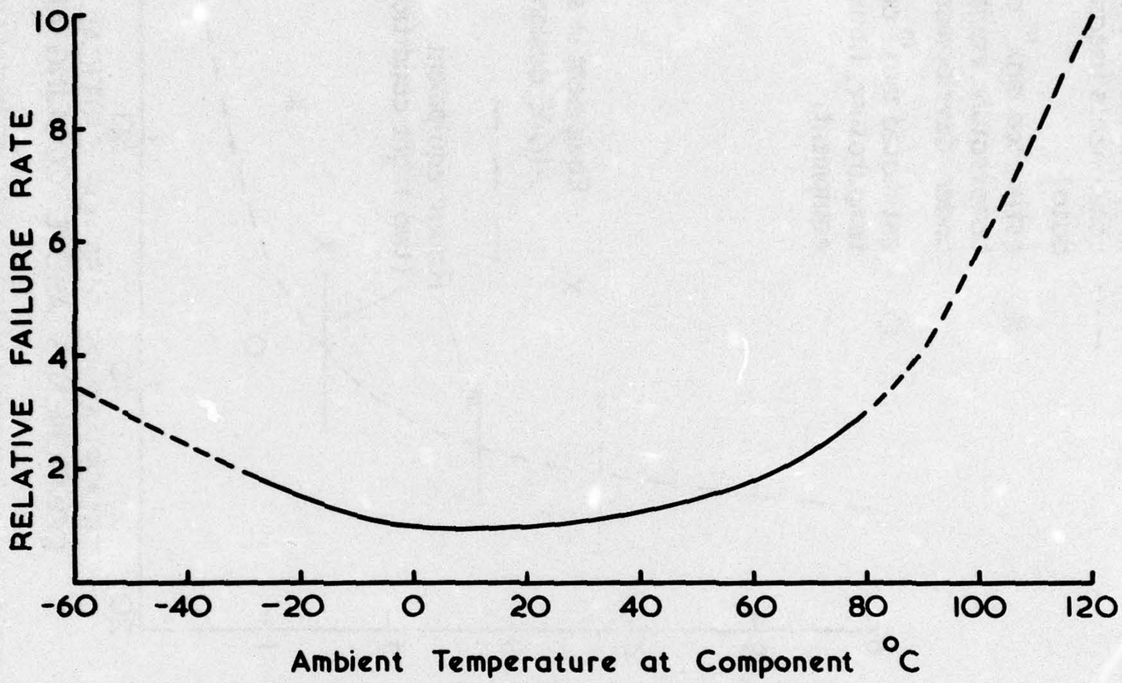

Figure 3 Relative Failure Rate with Temperature for Electronic Equipment.

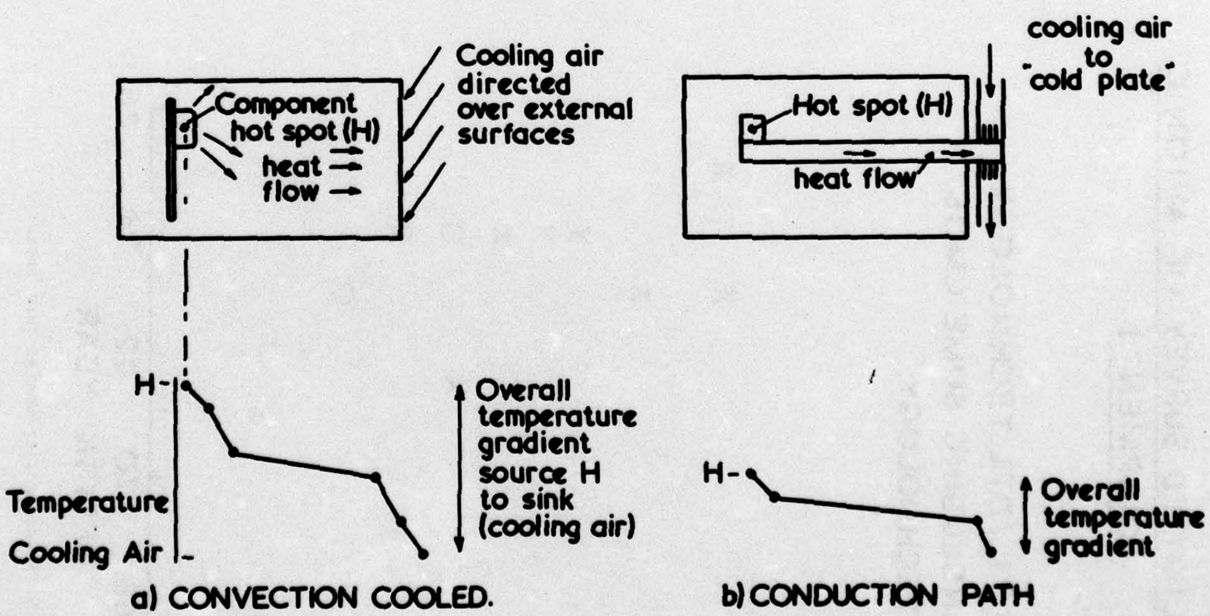

Figure 4 Thermal Performance of Two Typical Equipment Designs.

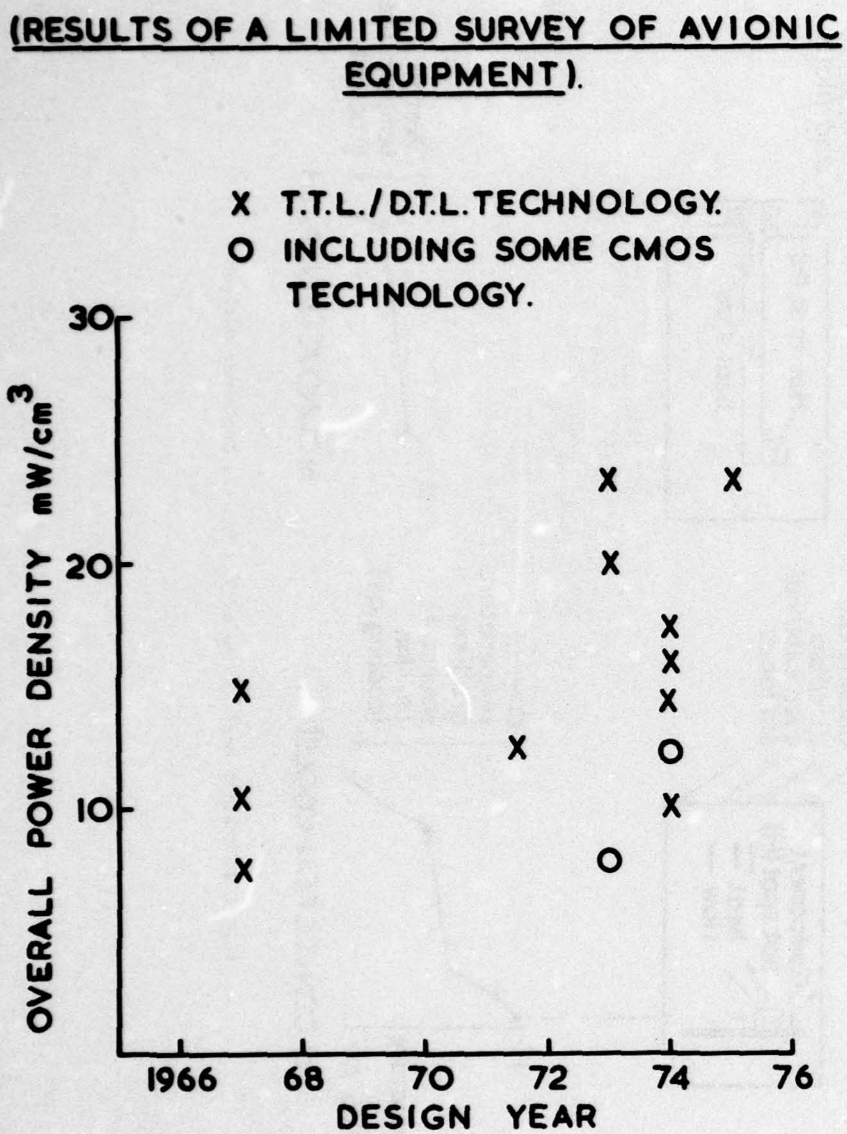

Figure 5 Power Density/Design Year Trends .

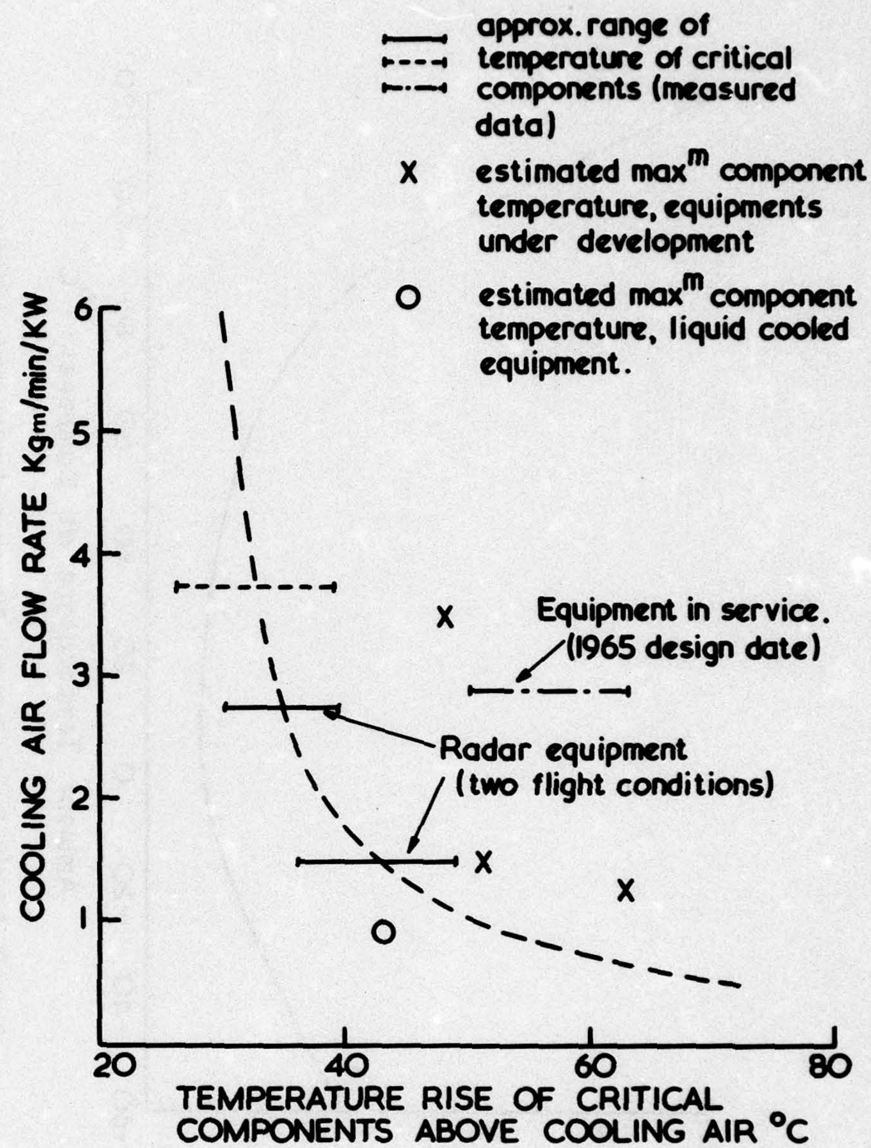

Figure 6 Cooling Performance of Several Radar and Avionic Equipments.

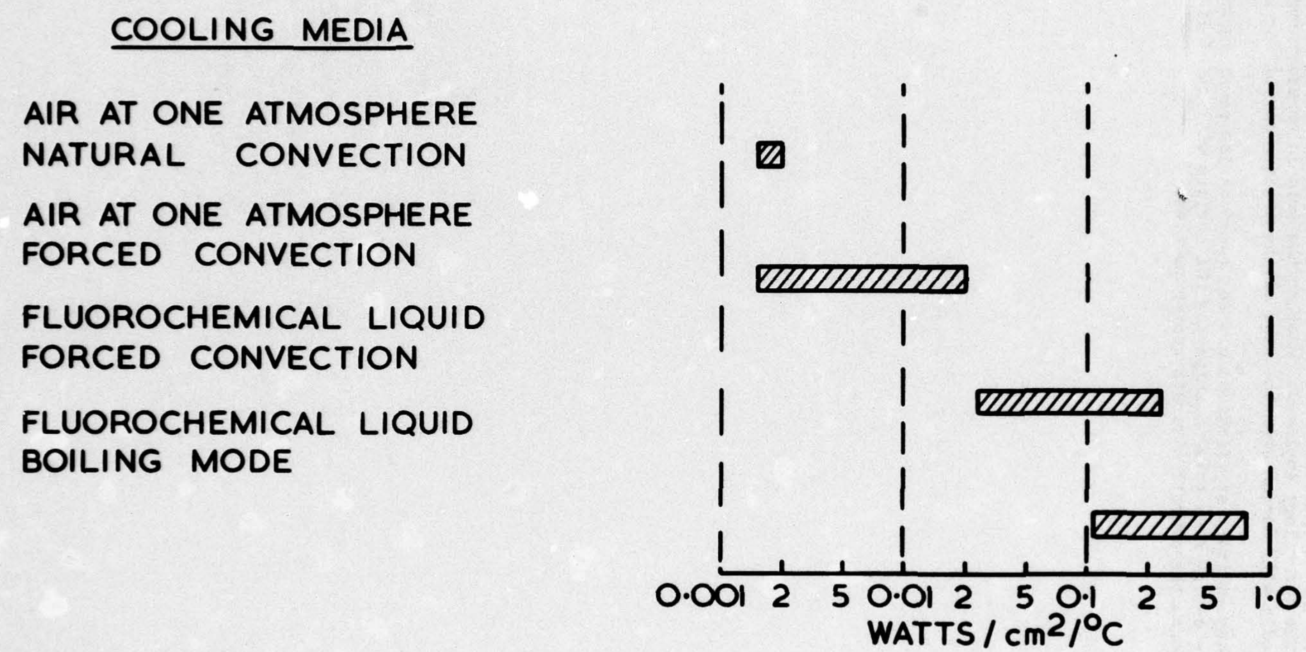

Figure 7 Comparison of Heat Transfer Co-efficients for Air & Fluorochemical Liquids.

DISCUSSION

N L Sigournay:

Large scale integration devices will always dissipate more power as the speed rises. Fundamentally the difficulty is that the semi-conductor manufacturer tries to get the best performance/yield he can, often resulting in high watts per square micron on the surface. Should Avionics manufacturers limit the semi-conductor manufacturer's chip dissipation?

G German:

The manufacturer of LSI is a subject outside my field. I understand however that the design of an LSI device is the result of a very complex compromise between a number of factors which includes chip size, yield, cost, performance and the technology employed. Whether the avionic manufacturers can exert a significant influence undoubtedly will depend upon the size of the avionic market for LSI devices.

One export view is that for many military applications devices based on low power technology will meet many requirements and special-to-type high power dissipating LSI devices will be of limited number not having a significant influence on the overall aircraft coding problem.

FINAL DISCUSSION SESSION

CHAIRMAN: Mr F S Stringer

PANEL MEMBERS: Mr H Timmers

Mr G R Giles

Mr K Morgan

M. P M A Desjean

Mr G C Letton Jnr

The Chairman opened the final session by thanking the audience for their excellent timekeeping which had made this specialist meeting so easy to run. He then went on to introduce the Panel who would assist him in providing the focus for this final discussion session.

He said that the lively question and answer periods following each of the formal sessions had indicated the interest in the subjects and that he had received a number of questions which he proposed to use as a basis for this final discussion period.

The first question from Mr Sigourny was in two parts:

a Is all of the equipment needed?

b Can some of the equipment be switched off some of the time?

Before opening this discussion from the floor he said his view was: no to the first, and yes to the second.

K Jackson

It seems to me that the need for equipment comes from the overall system effectiveness rather than the sub-system requirement. Therefore, it is something over which the equipment designer has very little control, it depends on the overall system effectiveness and cost.

I think you obviously can and sometimes should switch off, and that it ought to be recognised in the design of equipment and the overall system when it is desirable to do so.

K Morgan

The decision to switch something off is generally difficult to make. Before we decide that a system can be switched off in a compact military aircraft, we must look at all interfaces with the other pieces of equipment. For example, an arbitrary decision to switch off a particular piece of equipment could perhaps in effect switch off three quarters of the weapon system with which it interfaces.

J C Kweldam

The question of switching off equipment is one that has intrigued airline operators for some time. About 10 years ago we decided NOT to switch equipment OFF, but to leave it ON: we firmly believe that switching off equipment and switching it on later, does much more harm than leaving it on.

Whilst I have the microphone I would like to add that though we have talked a lot about the problem of cooling an electrical power arising from the increase of electronics, I have heard nobody suggesting that there might be other ways hydraulic, or pneumatic, of achieving the same end. Perhaps somebody should consider a trade-off study of alternative system and technical approaches.

Chairman

Its an interesting question, and I have one experience of that recently, when we were able to produce a hydraulic solution that was actually cheaper than the electronic solution. But before we had finished the work, the customer said he didn't want it.

G German

Another point about switching off loads is the difficulty of obtaining an overall benefit. For example if one switches off a transmitter and puts it on standby, and at the same time a computer processing information is left on, you cannot redirect the air from the transmitter to the computer you have left on, you still have air blowing through the transmitter. Switching off and redirecting the air would add considerably to the complexity of cooling systems.

K Morgan

There is another factor one needs to consider, if the equipment outside the equipment bay is switched off, the temperatures could easily drop below the dew point and there would be problems due to condensation.

K P Gerrity

If we were about to adopt a policy of switching off, then we would probably need a fairly complex, possibly multiplex system, to arrange for switching purposes, this in itself would lead to unreliability, and we may find we have traded off a cooling problem for another, resulting in even poorer reliability.

G Borgonovo

I am not convinced we could make a significant saving by switching off equipment. Except in special cases the equipment we could switch off would rarely exceed 10% of the total. Also, I am of the opinion that almost anything that can be done electronically is done electronically. It is clearly easier, for example, to cool a hydraulic switching mechanism on a camera which can tolerate temperatures up to 200°C than an equivalent electronic device which cannot tolerate a temperature of more than 65°C.

G C Letton Jnr

In designing the B1 we found that the total load with everything switched on was 180kw; but by studying the mission profile, it was shown that there was never need for more than 120kw of equipment to be switched on at any one time. Therefore the B1 was designed for 120kw load with a switching facility to switch off those loads which were not required.

G W Underwood

Since a major problem of cooling for light military aircraft concerns the cooling of cabin equipment whilst the aircraft is on the ground, I would like to ask if consideration could be given to the use of an auxillary power unit.

In the general discussion which followed it was thought that auxillary power units were not a practical proposition for military aircraft. For example, the provision of fuel for long periods of ground running would represent a major logistic problem.

Chairman

The next question I have for discussion comes from Mr Mehrtens and concerns the temperature of air provided for avionic cooling. Mr Mehrtens says that if cooling air at 35 to 37° is provided for cockpit cooling, why cannot we have very much cooler air provided to the equipment bay? He also makes the point that in fact when aircraft is on the ground in a hot climate, air is going into the equipment bay at 55°C. This could lead for example to the temperature in places such as a computer core store being as high as 85°C.

This is a situation which the equipment suppliers are saying is worse than is being suggested here, and represents a problem which can be extremely expensive to solve.

K Morgan

These problems have to be treated on a case by case basis. Modern military aircraft have to be capable of operation on a world wide basis, without the use of emergency power, or the use of cooling air supplied by external trolleys.

Returning to the question of 35°C inlet temperature, in the case of the European Environment and adding say 3° due to working on the air, one would expect to operate with a temperature of 30° right down to minus quantities, depending on where you are in Europe. However, world wide operation is likely to specify operations in 50°C, albeit, on a one chance in 10 years basis. The point I am making is that once the design requirements are agreed and accepted it is pointless asking for relaxation of the equipment specification on the grounds that these conditions occur infrequently.

A further problem in providing cooling air, is the deposition of moisture. Whilst it may be possible for the air to be cooled to as low as 10°C, cooling to this temperature could create more severe problems due to moisture in the air, and therefore any cooling requirement of this nature must be dealt with on a case by case basis.

CHAIRMAN

Clearly, the topic of cooling both the pilot and the equipment when the aircraft is on the ground is important. Since this relates to Mr Underwoods earlier discussion on auxilliary power units, I will ask him to start the discussion on this topic.

G W UNDERWOOD

Most military aircraft spend more time on standby on the ground than they do actually flying, and in an environment that is, in its overall effect, often more severe than the flying environment, with a consequent serious effect on reliability. The question is when to switch equipment on rather than switching it off when the aircraft is on the ground.

P M A Desjean

This also applies to civil aircraft, the airbus for example for every hour of flying spends one hour on the ground, in both cases with the equipment switched on.

G F Stevenson

You can provide fairly economical cooling on the ground. For example a system providing 6kw of cooling air in a flight regime of about Mark 1 at sea level could, with only a small depression of the outlet compressor of about 2 psi, meet the cooling requirements with only about 5 or 6 hp to provide this 6kw of cooling. In other words there are economical ways of providing cooling with ground equipment, using the same aircraft systems that are used in flight.

Chairman

I agree, but there are also requirements for hydraulics and electrics on the ground, and even though their requirements are also modest, this might lead to a large number of ground trucks.

G F Stevenson

I was only addressing the question of cooling, but of course a clutter of ground trucks would be a serious question, particularly for aircraft like the Harrier.

G German

Reliability is both temperature and time dependent, and whilst an excedence of a high temperature for a short time might exceed a specification, it might not be nearly as serious as a much less increase in temperature over a long period.

I would like to see specifications which spell out a temperature versus time profile. For example the often quoted 50°C of ground temperature only occurs at rare times in places like Death Valley in California.

H Timmers

Whilst this might be true for shade temperatures, some cockpit temperatures through perspex often exceed 50°C in temperature climates.

K Morgan

All of the considerations we are talking about, must be treated in the context of the directive given. We must consider the given directive, treat aircraft case by case and then try to put into practice what we are talking about today.

Chairman

We seem to have had a fairly full discussion on cooling problems. I would like us now to consider the question of possible use of high voltage DC power supplies for avionic equipment.

Newton

The suggestion of high voltage DC power supplies for systems of high power levels does not seem possible. I believe Mr Sigourny in his Paper was only talking about low power 3kw systems.

The possibility of obtaining a contractor to operate at 120v at anything up to 3 or 400 amp is well beyond the state of the art, let alone the problem raised by the necessity for contractor cooling.

Also I am not sure about the possibility of weight saving in the conductors, it seems to me that this Paper particular referred to another which considered 400v DC transmission systems rather than a proposed 120v DC system and was not comparing like with like.

Chairman

Since the future of high voltage systems does not appear very bright, perhaps we could turn now to the question of batteries. In saying this I am not thinking of their use in high power consuming systems such as radar, but rather the mass of low power consuming avionic equipment. Should not we be using batteries more for a lot of this solid state type of equipment and just changing the batteries over when they run out?

B E C Watt

I think that primary cells would run into the same kind of difficulties as secondary cells as far as temperature and environmental factors they can withstand. We are already speaking of operating temperatures being a problem in basic equipment. The aircraft secondary battery has a fairly big mass which protects it to a certain extent from rapid temperature changes, whereas I would expect small primary cells to be effected by such changes.

K P Gerrity

Two points concern me about the possible use of primary cells. Firstly, the energy density of batteries is low, and even though there has been some development over the last 10 years, they still represent a very uneconomical power source - not only in terms of cost, but also in terms of mass and volume. Typically, considering the 15 minutes standby loads with all radars and similar devices switched off, the secondary cells could weigh something like 35kgs. Secondly, batteries are notoriously difficult and expensive to maintain. If possible I would like to see batteries done away with.

P W SMITH

I believe one use for small batteries is to maintain power to special equipment such as computer stores when on the ground; also systems as inertial navigation and aircraft recording. I would also like to see the development of sensors which would give the state of the battery and thus relieve some of the maintenance problems.

H Timmers

I know of only one battery which will give power at above 70°C and have not found it possible to obtain a battery with a wide operating temperature range.

The Chairman then thanked all the participants for providing a lively and interesting discussion period and began the formal closing ceremony.

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14. Abstract

The continued increase in the quantity of avionics equipment in military aircraft has already given rise to a critical situation in terms of cooling. The environment, particularly at high speed and low levels, has made the use of the airframe or the fuel as a heat sink a less profitable arrangement than in the past. Alternative solutions must be found which include both the reduction in sources of heat and more efficient methods of cooling.

This specialist meeting was designed to familiarise the NATO scientists and engineers with current research and development work in the field of cooling and electrical power supplies, to define the problem quantitatively, and to identify areas which require research.

ABSTRACT

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